C. SPENNY

STUDY OF STABILIZATION, GUIDANCE AND CONTROL DESIGN CRITERIA DEFINITION

ITEM Ic, GUIDANCE & CONTROL SYSTEM MECHANIZATIONS AND TRADEOFFS

06314-6005-T001

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PREPARED FOR NASA ELECTRONICS RESEARCH CENTER

UNDER CONTRACT NAS 12-110

TRW SYSTEMS

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TRW SYSTEMS

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Redondo Beach, California

FOREWORD

This report presents the results of Item 1c of TRW's study under NASA-ERC contract NAS 12-110 (A Study of Stabilization, Guidance & Control Design Criteria Definition). System mechanization genealogies were developed to support the SG&C Design Criteria outline. Plausible Guidance and Control Configurations were postulated and the characteristics of each mechanization were discussed. A brief survey of typical sensors, as used for Guidance and Control functions was included.

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1.0 INTRODUCTION

Since the earliest satellite launching, there have been probably as many distinct space guidance and control mechanizations developed as types of space vehicles. This is true in large part because the selection of the G&C configuration ultimately depends so heavily on the particular space vehicle and its intended mission. Recent studies, notably the AF sponsored Standardized Space Guidance Study, have been conducted with the objective of developing a standardized guidance system concept and mechanization for a variety of near future space programs. These studies, conducted with certain constraining ground rules (e.g., the use of a specific launch vehicle), have shown this approach to be both feasible and economically attractive.

However, at this point in time, such a statement cannot be conclusively made about space missions contemplated for the next two decades and which were not considered in the previous studies. It is the intent in this study sub-task to configure, without regard to standardization, alternative G and C mechanizations which would meet the anticipated functional requirements of future space missions.

The specific goals were to:

1. Postulate plausible Guidance & Control (G&C) configurations
2. Prepare a G&C system genealogy which will provide a logical organization structure for the G&C mechanization
3. Consider gross characteristics of the mechanization classes.

The general objective of this subtask was to support the formulation of the SG&C Design Criteria outline. The genealogy will be used as an aid in the classification of Design Criteria subjects. The configuration descriptions and characteristics will be employed for the contents definition of the monographs.

It should be understood in the discussions to follow that the phrase "guidance and control" is meant to encompass the broader functions of navigation and stabilization as well. The guidance functions then include

1. Navigation, or the determination of the vehicle trajectory state based on onboard and/or ground tracking measurement data
2. Targeting, or the determination of a desired end state
3. Guidance computation, or the determination of the trajectory perturbation required to ensure that the desired end state can be achieved in an optimum way
4. Command generation, or the determination of vehicle steering and engine thrust commands required to effect the proper trajectory corrections.

The control functions include

1. Vehicle stabilization, or the maintenance of the spacecraft in a known and desired attitude
2. Maneuver control of the vehicle from one orientation to another as required for purposes of trajectory correction, navigation, experiments or other spacecraft functions. The control functions may also include engine thrust control.

The targeting, guidance computation, and command generation functions more properly fall into the area of guidance system software development and will not be discussed in detail in the sections to follow. The primary emphasis is on system configurations and mechanizations which when properly utilized provide sufficient data that these functions can be performed.

Section 2 presents a summary of conclusions. In Sections 3 and 4, G&C configurations are postulated and mechanization characteristics are discussed.

2.0 SUMMARY

- A large variety of G&C system configurations can be formulated. The postulated systems were limited to the most realistic and practical concepts, and resulted for the Guidance and Control systems in respectively 20 and 14 different configurations.

- The G&C systems genealogy was developed to a degree satisfactory for the SG&C Design Criteria outline, and is presented in Section 3.1 and 4.2.

- The Guidance systems genealogy was based on the differentiation between radio guided and autonomous concepts. The latter ones were divided in all inertial, aided inertial and all optical systems.

- The Control systems genealogy was based on a differentiation between passive, active and semi-active concepts.

- The gross functional characteristics of the different classes of systems were discussed. The physical characteristics will depend on the specific applications and can vary for each over a wide range. This holds particularly for the control systems.

3.0 GUIDANCE SYSTEM CONFIGURATIONS

3.1 Guidance System Types and Genealogy

A basic block diagram of a generic guidance and control system, representative of either an autonomous system, a radio controlled system, or a combination of the two is illustrated in Figure 3.1. The navigation function can be performed using onboard inertial sensors, with or without auxiliary electro-optical or electro-magnetic devices, ground radar tracking or a combination of both. The targeting, guidance computation, and command generation functions can be performed either in an onboard computer or, for the case where the navigation function is performed via ground radar tracking, a ground based computer installation.

The ground controlled, or radio guidance, system can be of two forms - closed loop or open loop guidance. In closed loop radio guidance, as typified in the earlier missile launch guidance systems (e.g., Atlas), vehicle steering corrections are continually commanded through the control system on the basis of continually updated navigation and guidance computation information until thrust termination. The thrust termination is likewise determined on the basis of continuous navigation and guidance data. In open loop radio guidance, as typified in midcourse correction schemes for lunar and interplanetary probes (e.g., Ranger and Mariner), radio derived navigation data is used to determine S/C (spacecraft) orbit, consequent target miss, and the necessary velocity vector increment required to minimize this miss. This correction is transmitted to the S/C control system as S/C attitude reorientation commands and some form of velocity magnitude command (e.g., thrust-time, acceleration-time, or simply ΔV). The commands are then effected open loop and subsequent corrections may be commanded on the basis of continued tracking.

The basic types of guidance schemes mentioned above and the various classifications of these types are illustrated in the genealogy charts, Figure 3.2 through 3.6. The major breakdown shown is between autonomous and radio guidance

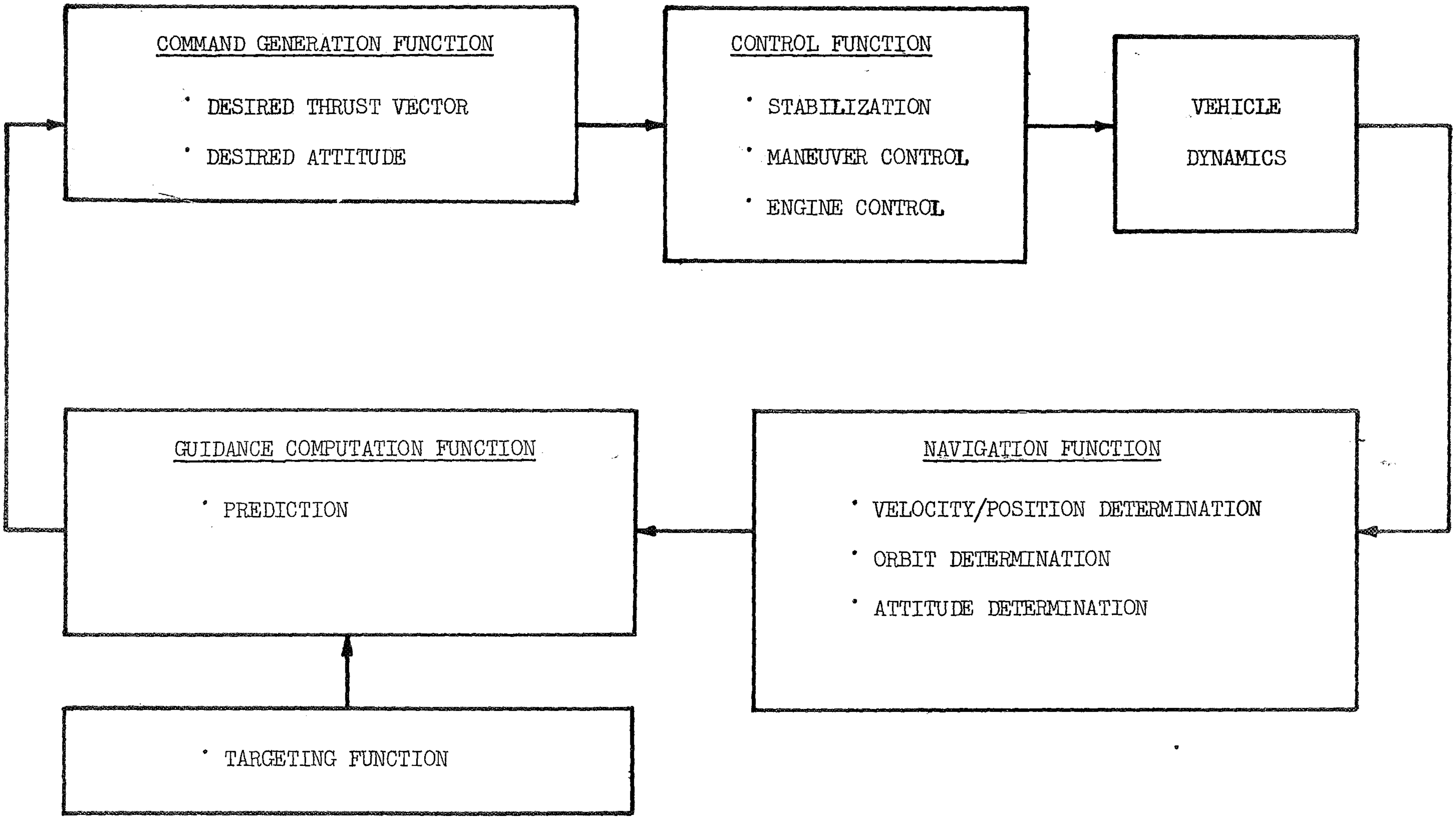

FIGURE 3.1 GENERIC GUIDANCE AND CONTROL SYSTEM

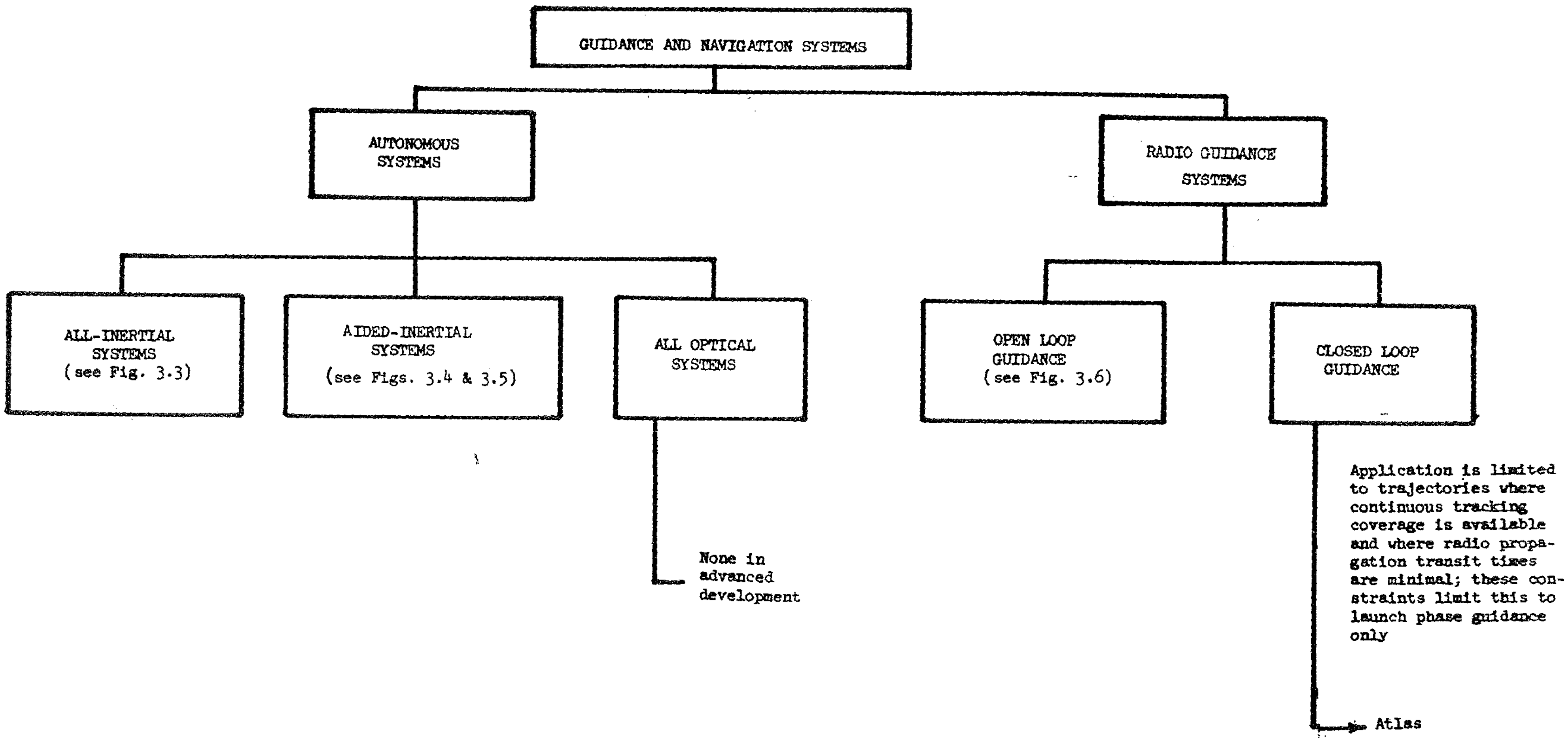

Figure 3.2. Major Guidance and Navigation Systems Genealogy Breakdown

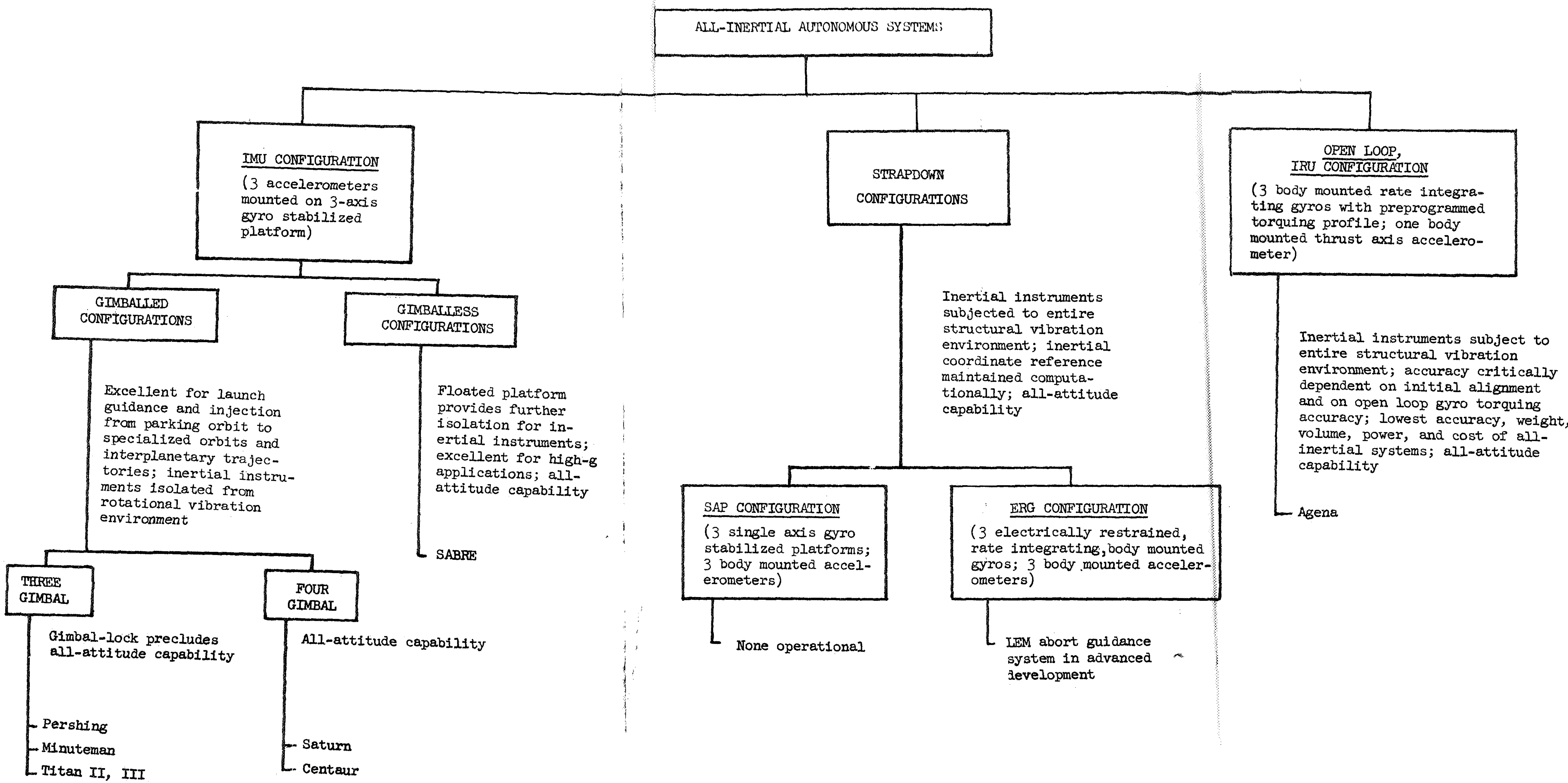

Figure 3.3. Genealogy Breakdown of All-Inertial Autonomous Guidance Systems

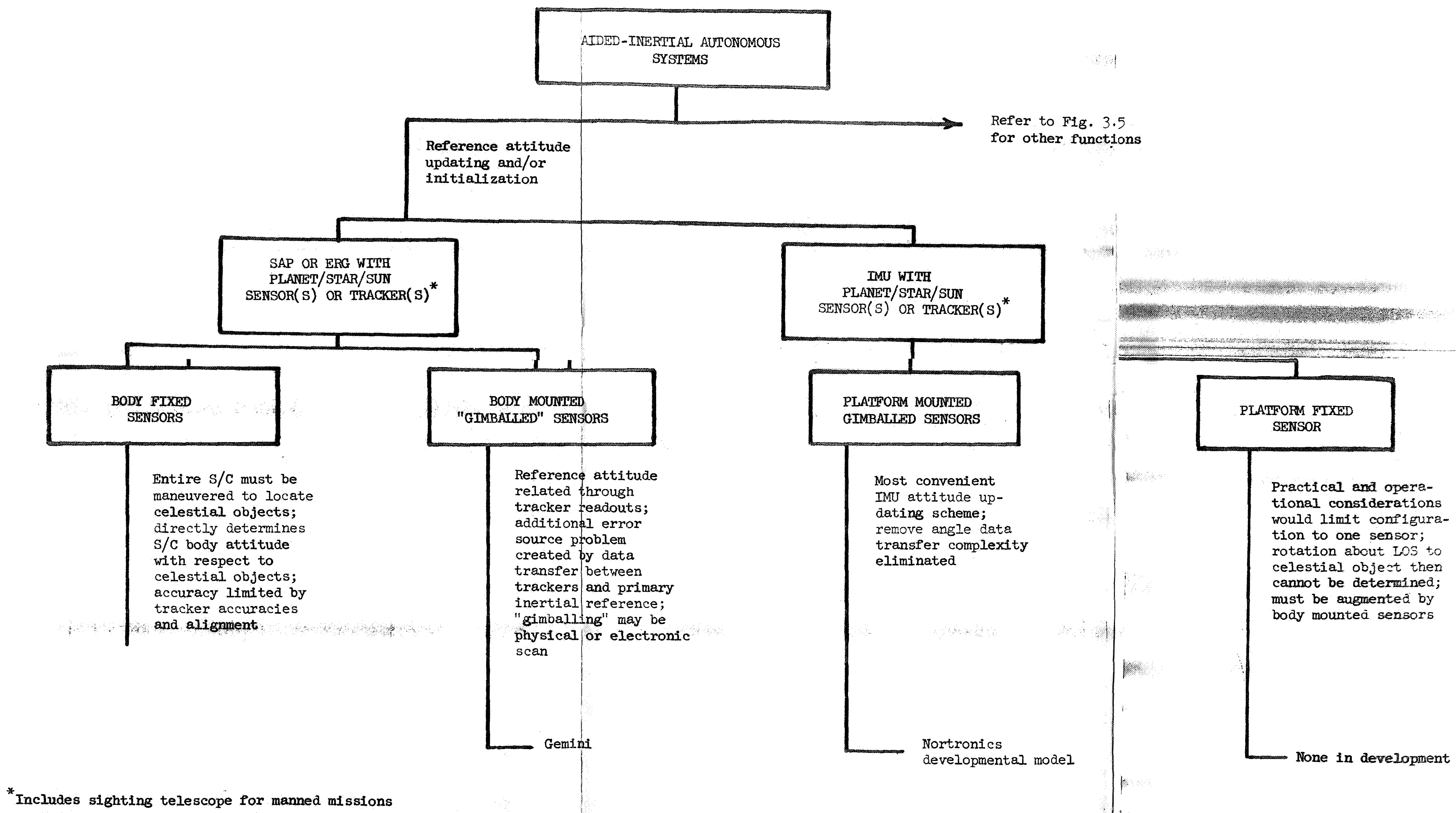

Figure 3.4. Genealogy Breakdown for Aided Inertial Autonomous Guidance Configurations (Part A)

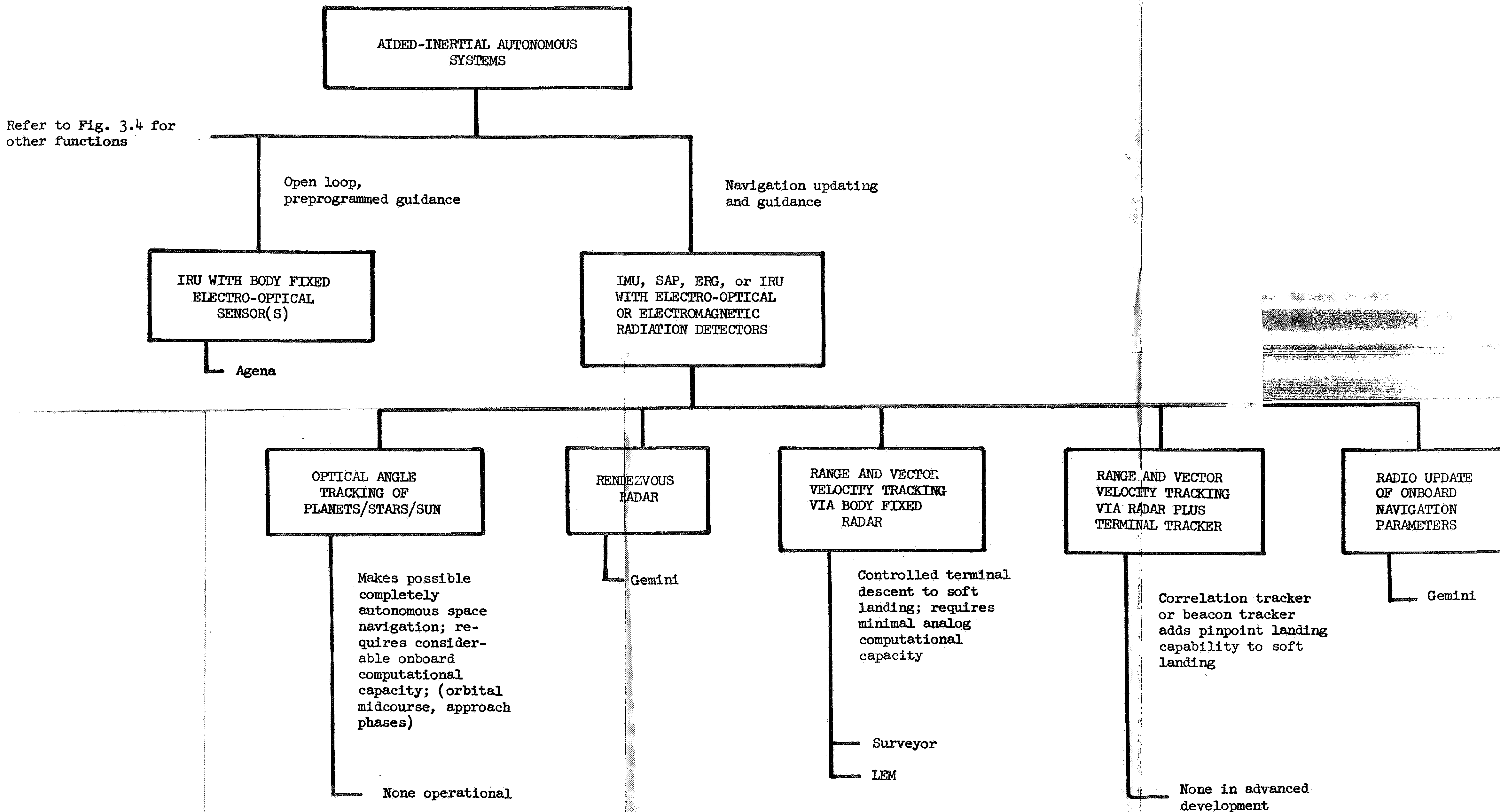

Figure 3.5. Genealogy Breakdown for Aided Inertial Autonomous Guidance Configurations (Part B)

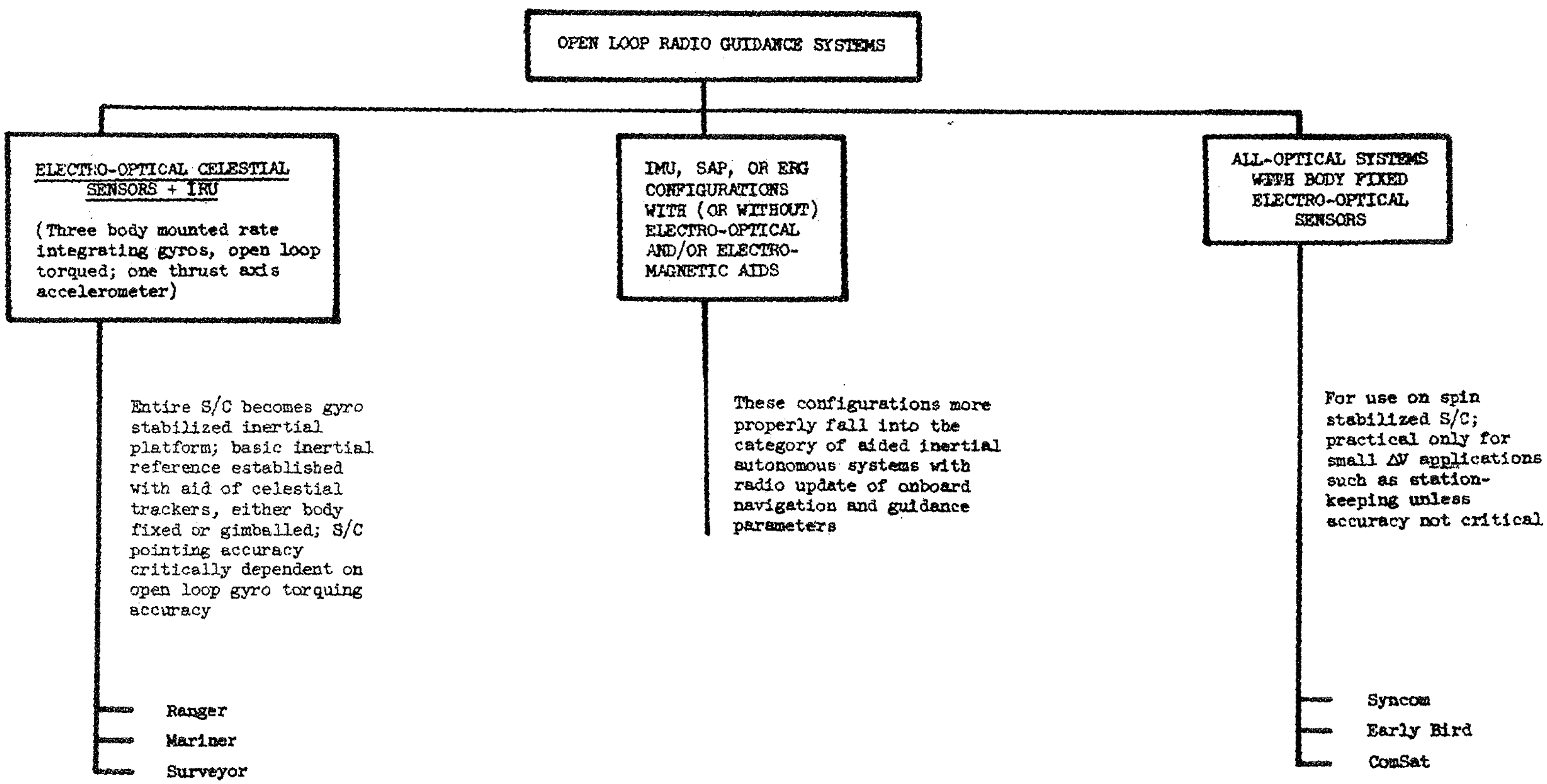

Figure 3.6. Genealogy Breakdown for Open Loop Radio Guidance Systems

systems. The combination radio-inertial system type as typified by Atlas/ Centaur is not included in the chart since in effect such systems are really purely radio guided or purely autonomous during specific mission phases. It is also noted that several of the example S/C cited in the charts appear under more than one classification. This is a reflection of the changes in system configuration and functional operating modes required for different mission phases of a given S/C.

A breakdown of the all-inertial autonomous systems is shown in Figure 3.3. In general, these configurations are applicable primarily for launch phase guidance. The aided inertial autonomous systems are utilized in several modes. In the first (Figure 3.4), the optical aids are used for reference attitude updating and/or initialization. With a known reference attitude established, electro-optical or electro-magnetic sensors can be used for autonomous navigation (Figure 3.5). A general configuration for a large portion of space missions will be the open-loop radio guided system, Figure 3.6.

3.2 Representative System Configurations

A variety of guidance system configurations can be contrived. In Table 3.1 an attempt has been made to list a reasonable number of configurations which would be representative of future systems. All the listed configurations either are, or are closely akin to, systems in operational usage or in advanced stages of development today.

It should be emphasized that in the interests of maintaining a reasonable listing of configurations, only broad configuration classifications have been given in Table 3.1 with no attempt at detailed breakdowns. For example, those configurations listed as utilizing star sensors could be further broken down into configurations with

1. Body fixed star sensors

2. Body mounted, gimballed star sensors

TABLE 3.1 REPRESENTATIVE GUIDANCE SYSTEM CONFIGURATIONS

MISSION PHASE			LAUNCH AND EARTH ORBIT INJECTION				EARTH ORBIT CORRECTION OPERATIONS, AND TRANSFERS									COAST AND MIDCOURSE CORRECTIONS			PLANETARY APPROACH			TERMINAL DESCENT						
CONFIGURATION			1	2	3	4	5	6	7	8	9	10 i	10 ii	10 iii	10 iv	11	6'	10'	11'	12	13	14	15	16	17	18	19	20
TYPE OF GUIDANCE SYSTEM	AUTONOMOUS		X	X	X						X			g			g	g	g	X	X	X	X	X	X	X	X	X
	RADIO, CLOSED LOOP					X																						
	RADIO, OPEN LOOP						X	X	X	X		X				X	X	X	X									
GROUND BASED RADAR SYSTEMS AND COMPUTER						X	X	X	X	X		X				X	X	X	X									
VEHICLE MOUNTED EQUIPMENT	INERTIAL	IMU (a)	X																									
		SAP (b)		X							X	X						X			X				X	X	X	X
		STRAPDOWN ERG (c)			X																							
		STRAPDOWN IRU (d)				X		X	X								X			X			X	X				
		RATE GYRO	e	e	e	e		e	e	e	e	e					e	e		e	e			e	e	e	e	e
	ELECTRO-OPTICAL AND ELECTRO-MAGNETIC RADIATION DETECTORS	STAR SENSOR OR TRACKER						X	X	X		f					X	f		X	f							
		SUN SENSOR					X	X				f				X	X	f	X	X	f							
		PLANET SCANNER OR TRACKER					X		X	X		f				X			X	X	X				X			
		SPACE SEXTANT OR SIGHTING TELESCOPE												j			j	j		j	j							X
		V/H METER																							X			
		CORRELATION TRACKER											h				h	h		h	h					X		
		BEACON TRACKER																									X	
		LASER RANGER													f									f	f			
		RANGE RADAR													f							X	X	f	f	X	X	X
		DOPPLER RADAR													X									X	f	X	X	X
	COMPUTER		X	X	X						X		X	X	X					X	X				X	X	X	X
	PROGRAMMER/SEQUENCER					X	X	X	X	X		X				X	X	X	X			X	X	X				

a. IMU = INERTIAL MEASUREMENT UNIT = 3 OR 4 GIMBAL GYRO STABILIZED PLATFORM MOUNTING 3 MUTUALLY ORTHOGONAL ACCELEROMETERS
b. SAP = SINGLE AXIS PLATFORM; PLATFORM INPUT AXIS SIGNALS PROCESSED BY COMPUTER TO DETERMINE S/C BODY AXES/INERTIAL REFERENCE AXES RELATIONSHIP; BODY MOUNTED ACCELEROMETERS
c. ERG = ELECTRICALLY RESTRAINED GYRO OF RATE INTEGRATING TYPE; GYRO TORQUING SIGNALS PROCESSED BY COMPUTER TO DETERMINE COORDINATE RELATIONSHIP; BODY MOUNTED ACCELEROMETERS
d. IRU = INERTIAL REFERENCE UNIT; FOR CONFIGURATION 4, CONSISTS OF VERTICAL AND HEADING REFERENCE; FOR ALL OTHER CONFIGURATIONS, CONSISTS OF BODY MOUNTED RATE INTERGRATING GYROS WITH BASICALLY ONLY AN OPEN LOOP TORQUING CAPABILITY AND USUALLY ONLY A SINGLE BODY MOUNTED ACCELEROMETER WITH INPUT AXIS ALONG FIXED ENGINE THRUST LINE
e. RATE GYROS CAN BE USED FOR STABILIZATION BUT NOT ABSOLUTELY ESSENTIAL
f. INDICATES EITHER OR BOTH TYPES OF EQUIPMENT CAN BE USED
g. INDICATES THAT GIVEN SUFFICIENT ONBOARD COMPUTATIONAL CAPABILITY, SYSTEM CAN BE MADE AUTONOMOUS
h. INDICATES STAR FIELD MAPPER; CAN BE USED TO AID STAR ACQUISITION OR TO PROVIDE AS MUCH AND MORE DATA THAN IS OBTAINABLE FROM STAR AND SUN SENSORS
j. FOR MANNED MISSIONS

3. IMU mounted, fixed star sensors (IMU = Inertial Measurement Unit)

4. IMU mounted, gimballed star sensors

The gimballed star sensors could be further broken down into those having either one rotational degree of freedom or two degrees of freedom. Further categorization could include such details as method of scanning and tracking. However, for this discussion, it is felt that such fine classification is neither necessary nor warranted.

Configuration No. 1 (Conventional Inertial Guidance Platform)

This autonomous system, with a fully gimballed IMU has been used with notable success in present day missiles and space boosters. However, such factors as performance limitations imposed by long term gyro drift will restrict such a configuration to usage mainly for the launch and ascent phases of future space missions. Further discussions of the autonomous IMU configuration will be found under Configuration Number 9 and in the following paragraph.

Configurations No. 2. & 3 (Strapdown)

In Configuration No. 2, the three axis gyro stabilized platform is replaced by three single axis platforms (SAP's), and the relationship between the S/C body axes and a reference inertial coordinate system is computed rather than physically maintained by gimballing. The computation is based on platform input axis signals, from each of the SAP's, operated on by a suitable algorithm. The most commonly used algorithm results in the continuous updating, either by a DDA* or a general purpose digital computer, of the direction cosines between S/C body and inertial reference axes.

*DDA = Digital Differential Analyser

In Configuration No. 3, body mounted, electrically restrained gyros (ERG's) of the rate integrating type are used. The gyro torquing signals provide the computer with the data necessary for S/C body inertial reference coordinate updating.

Configurations 2 and 3 have not been operationally demonstrated as yet for any space missions. However, Configuration 3 is now in an advanced state of development at TRW Systems for the LEM AGS and scheduled for first flight testing in 1967.

It is to be noted that the IMU, SAP, and ERG configurations, although mechanized differently, provide the same basic navigation data - S/C velocity and position, as well as attitude, in a known inertial coordinate reference frame. Because of the functional similarity of these inertial configurations, they have been lumped together for purposes of discussing the remaining configurations. When discussing specific missions and interfaces between other equipment however, it should be remembered that the IMU, SAP, and ERG present altogether different problems.

Some preliminary assessments of the relative merits of these three configurations can be made based on past experience and the data presented in Section 3.3. The system with the most potential accuracy is configuration No. 1 with the complete IMU. At the same time however, this high accuracy configuration will tend to be the heaviest of the three.

The strapdown configurations are inherently more inaccurate than the IMU configuration, primarily because of the full vibration environment that the strapdown inertial instruments are exposed to. The strapdown configurations can offer the advantages of being lighter, physically less complex, and potentially more reliable than an IMU configuration. Of the two strapdown configurations, the ERG configuration should be lighter and more reliable than the SAP configuration. For the IMU configuration the capability of arbitrary (limited by gimbal freedom) platform orientation allows relatively simple instrument calibration while the IMU is installed in the S/C. Similar calibration of the strapdown configurations generally involves more complex operations and less calibration accuracy. Initial alignment and leveling of the three configurations are comparable; in the IMU case, physical alignment and leveling is involved whereas for the strapdown configurations, the process is computational.

Configuration No. 4 (Closed Loop Radio Guidance)

This is the configuration used in the early ballistic missile programs (e.g., Atlas). The term IRU (Inertial Reference Unit) in this case indicates simply a heading and vertical reference whose output is differenced against commands generated on and transmitted from the ground to provide steering signals to the flight control system. The engine staging signals are also ground computed.

This configuration naturally requires a near minimum of inertial equipment. However, since closed loop guidance and control is effected via radar tracking-ground computation-radio transmission, the inherent transit path delays restrict its usage to near-earth missions, primarily for launch guidance applications. The guidance and control functions with this configuration are also constrained to only those portions of the trajectory where tracking coverage is available.

Because of the above limitations, Configuration No. 4 is the only one labeled as a closed radio guidance system. All other radio guidance configuration in Table 3.1 are of the open loop type.

Configuration No. 5 (For Near Earth, Spin Stabilized S/C)

With body fixed, passive electro-optical sensors, this configuration is the simplest of any listed but is applicable only for spin stabilized S/C requiring only small velocity increments, ΔV, for minor orbit corrections, such as station keeping. The navigation and guidance function is provided through the ground tracking and computation net. The two optical sensors provide only S/C attitude data to the ground computer.

The optical sensors include a planet tracker and a sun seeker. For earth orbiting satellites on which it is desired to maintain the spin axis normal to the orbit plane, the sun sensor is not required after the S/C has been oriented approximately to the desired attitude.

There are some obvious limitations to the usage of this configuration. The primary ones are that complete inertial attitude determination can only be made when the sun is visible from the S/C location, the sun polar angle is not near zero, and the spin axis orientation is such that the earth IR image cuts across both earth sensor fields of view. Also, the accuracy with which the spin axis orientation can be placed in an arbitrarily desired attitude is on the order of one degree (3σ), thus precluding efficient application of large ΔV changes.

Configuration No. 6 (Surveyor Configuration)

The IRU (Inertial Reference Unit) in this configuration consists of three rate integrating gyros, body mounted in a mutually orthogonal manner. With proper attitude control system design, the entire S/C becomes a gyro stabilized platform, and the S/C orientation will tend to remain fixed in inertial space. Since no computational updating of the S/C orientation relative to inertial space is performed, some external data source is required to determine what this fixed attitude is. A proven and straightforward means of determining the S/C attitude is to use error signals from body fixed sun and star sensors to drive, through the attitude control system, the S/C to a known orientation with respect to the sun and a chosen star (e.g., Ranger, Mariner, Surveyor). Subsequent desired attitude changes as determined from ground computations can then be achieved by open loop torquing the appropriate gyros in the proper sequence. The alignment of the star sensor relative to the S/C axes and the sun sensor is somewhat mission trajectory dependent as is the choice of the reference star. Periods of star occultation and eclipses of the sun will render the sensors useless but during these periods, the attitude control can be closed through the gyro loops.

For ΔV vector application for orbital changes, the engine burn is controlled on the basis of ground computed commands in the form of either thrust versus time, thrust acceleration versus time or ΔV magnitude command. A single accelerometer with its input axis along the fixed engine nominal thrust line is the only required instrumentation. In some instances, especially where engine misalignments can be large, it is desirable to mount two orthogonal accelerometers with input axes normal to the nominal engine thrust line to help compensate for engine misalignments.

A slight modification to the above configuration would be to use gimballed sun and star trackers. The initial sun and star acquisition would have to be performed just as if the sensors were body fixed. However, the use of gimballed trackers provides a means for verifying that the open loop attitude change commands were implemented correctly. Any deviations in the measured tracker gimbal angles from the predicted values can be construed as an error in S/C attitude orientation and corrections can be made accordingly. The use of multiple star tracker units (e.g., OAO) can improve S/C pointing accuracy.

Configuration No. 7 (Earth-Star References)

This configuration is similar to Configuration 6, the only difference being the substitution of a horizon tracker for the sun sensor. For earth orbital mission, Configuration 7 will in many instances prove to have definite advantages, especially if the earth satellite is to have an earth pointing mode. The IRU is included to provide reorientation capability for small ΔV orbital correction maneuvers.

Configuration No. 8 (All-Optical Reference)

This is a modification of Configuration 7 with the IRU removed and two axis gimballing freedom provided for both the horizon tracker and the star sensor. This modification trades the increased complexity of the sensor gimballing for the IRU long term reliability. Note that a similar modification can be made to many of the subsequent configurations. However, again in the interests of maintaining a reasonable size listing, these related variations are not carried through the rest of Table 3.1.

Configuration No. 9 (All-Inertial Reference)

This configuration, while demonstrated to be a highly successful one for launch and ascent guidance, has definite limitations which will probably preclude its use, in the given form, for most other applications. Two of the more important considerations are

1. Gyro drift: For extended missions, a means of updating the inertial reference must be devised to compensate for gyro drift effects.

2. Energy management: Again for extended missions, it may be desirable to shut down the gyro operation for reasons of power conservation or thermal control. A means of re-establishing the inertial reference is required.

The required re-establishment and updating of the inertial refererence can be accomplished with the aid of celestial trackers, as is done in Configuration No. 10.

Configuration No. 10 (Aided-Inertial Reference)

Earth, sun and star trackers are indicated in Table 3.1 as those sensors required for inertial reference updating and reinitializing. The sensors used and the manner of implementation are mission dependent as summarized below.

1. Short term (days) earth orbit: For missions short enough that inertial reference shutdown is not warranted but too long to ignore gyro drift, a simple updating procedure can be mechanized using gimballed star trackers. On the basis of the nominal reference inertial coordinate directions and a chosen star, star tracker pointing angles relative to these directions can be computed and commanded. If the tracker has an acquisition FOV (field of view) of α deg, where α is some percentage larger than the maximum expected gyro drift, then the chosen star should be in the tracker FOV. The deviation of the star position from the tracker line-of-sight gives a direct indication of the amount the gyros have drifted since the last fix. Either one tracker, sequentially sighting on stars, or two trackers for simultaneous sighting can be implemented.

2. Long term (weeks and up) earth orbits: where a definite need exists for inertial reference shutdown, the problem of initial acquisition of the reference star(s) becomes a critical one. The problem arises because the nominal inertial directions used as a reference, as in the updating procedure above, are no longer available. Several methods are available for roughly establishing this nominal for coarse alignment preparatory to fine alignment, again somewhat dependent on mission.

   a. Unmanned earth orbit mission: For earth orbits which the orbit normal is not near the sun LOS*, knowledge of orbit normal direction, sun line and sidereal time is sufficient to establish a reference frame in which to locate the stars for coarse telescope pointing. Location of two stars in this manner then establishes a precise inertial frame. A sun sensor is required if orbital position is not known.

      A second scheme utilizes an earth horizon scanner and requires that the IMU, or its equivalent, be turned on prior to star acquisition. The procedure is to use local vertical information from the horizon scanner and to use the IMU in gyrocompassing mode to establish a space fixed coordinate system.

      A third scheme does not require accurate preknowledge of orbital characteristics. The reference inertial directions are established by the sun/bright star acquisition scheme discussed under Configuration No. 6.

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* LOS = Line Of Sight

In the first two schemes above, a correlation tracker, or star field map matcher could be used to aid star acquisition; alternatively, the star mapper can be used to provide all the information necessary to determine spatial reference directions.

b. Manned earth orbit missions: With the IMU, or its equivalent, operating in the inertial mode and controlling the S/C attitude, reference stars can be located accurately by the crew using either a simple sighting telescope or a space sextant. The measured star LOS angles relative to the presently maintained inertial directions help to determine the orientation of these directions.

The last sub-configuration shown under Configuration No. 10 includes a ranging device and a doppler radar. These units add rendezvous capability to a basic configuration. Although this capability is shown listed under Configuration No. 10, it can be provided as well for Configuration No. 6.

The S/C with rendezvous capability is a prime example of a S/C with combined autonomous/radio guidance capability. Radio guidance is necessary to vector the S/C to a point where onboard rendezvous trackers can acquire the target S/C. Following acquisition, tracker directed autonomous guidance would be used.

Configuration No. 11 (For Spin Stabilized S/C Far Removed From Planets)

This configuration is basically the same as that of Configuration No. 5. The difference is that the earth target shape has changed from a disc to more nearly a point source as S/C to earth range increases. The critical design area for this configuration is the problem of determining at what ranges from the target planet this scanning method can be used.

Configuration No. 6' and 10' (Midcourse Configurations for Three Axis Stabilized S/C)

These are essentially the same as Configurations 6 and 10. The primary mode of guidance would be based on ground tracking data. However, data from celestial body sightings can be combined with the tracking data to obtain the best estimate of the S/C trajectory. In the extreme, for manned missions, guidance could be made completely autonomous by relying solely on a history of celestial sightings and using this sighting data for orbit determination, e.g., via Kalman filtering techniques.

Configuration No. 11' (Approach Guidance Configuration for Spin Stabilized S/C)

This configuration can be considered autonomous only in the sense that the planetary approach phase, beginning at about 100 planet diameters, could be made autonomous, given sufficient computational capability. All phases leading up to the planetary approach phase (except for launch injection) would most practically be radio guided for the specific case of a spin stabilized S/C.

At 100 planet diameters, the planet subtense is approximately 0.5°. At this angle subtense, the planet image pulse width in a field of view can be used as a measure of apparent diameter for purposes of stadimetric ranging. This data, along with sun and planet angular position data (i.e., the polar angle information discussed under Configuration No. 5) provides sufficient information for navigation relative to the target planet.

Configuration No. 12 and 13 (Approach Guidance Configuration for Three Axis Stabilized S/C)

Again these configurations are essentially the same as Configurations 6 and 10. Here however, the planet scanner is an important addition to provide both stadimetric ranging data and planet LOS (Line of Sight) data.

Configuration No. 14 (Open Loop, Semi-Soft Landing for Spin Stabilized S/C)

This and the subsequent terminal descent configurations are all listed as autonomous. It is to be remembered that these configurations are autonomous only in the sense of this mission phase. The guidance functions during most of the mission phases leading up to terminal descent (except launch) may have been radio controlled.

It should also be noted that for all the terminal descent configurations, a preparatory phase in which the S/C attitude is controlled to some desired approach orientation will be incorporated. For aerodynamic braking entry, the proper orientation is dictated by considerations of deceleration g levels, heating and heating rates. For propulsive deceleration, the attitude will most usually be selected so that the retro-engine thrust is nearly anti-parallel to the approach velocity vector. In either case, the intelligence for determining the proper attitude is obtained from basic approach orbit data and celestial or inertial references. The specific configurations required for this initial alignment phase are no different from any of the Configurations 5 through 13 and will not be discussed further. The subsequent discussions will be limited to the terminal descent phase proper.

Configuration 14 is the simplest configuration listed and is applicable only for spin stabilized instrument capsules with the retro thrust axis coincident with the spin axis. With the spin axis properly oriented during the pre-descent phase, an altitude-marking radar signals retro-engine ignition at some desired altitude. The ignition altitude is a predeterminable function of approach velocity, initial capsule weight, engine thrust and I_{SP} characteristics, and planetary atmosphere if any, and is chosen to give a semi-soft landing with impact velocities in the range of 20 to 200 fps.

Configuration No. 15 (Open Loop, Semi-Soft Landing for Three Axis Stabilized S/C)

This configuration differs from the previous one only in that it is applicable to a three axis stabilized landing capsule rather than a spin stabilized capsule.

Configuration No. 16 (Terminal Guidance Control for Soft Landing)

This is the first configuration that allows for true soft landing with impact velocities in the range of 10 to 20 fps. The initial portion of the terminal descent is similar to that for Configuration 15, with an altitude marking radar triggering main retro ignition. Main retro burnout occurs at an altitude and velocity where onboard range and velocity trackers can be used to provide closed loop descent trajectory control via vernier engines to a soft landing (e.g., Surveyor). The Surveyor and Apollo LEM use as the basic descent control sensors radar altimeter and doppler velocity sensors (RADVS). The listing in Table 3.1 shows an alternative range sensor - a laser ranger - which can give more range accuracy.

Configuration No. 17 (Terminal Guidance Control For Soft Landing)

The basic operation with this configuration is the same as with Configuration 16. However, the sensors used to determine range and velocity are different, involving a V/H meter to determine velocity to altitude ratio. For satisfactory descent control, some means of extracting velocity and altitude (or slant range) as individual parameters must be provided. This can be done by using either (1) some sort of ranging device or (2) a doppler velocity sensor. In either case, it will become obvious that the overall configuration is more complex than that indicated for Configuration 16, particularly since a planet tracker may have to be incorporated to point the V/H optical axis.

Configuration No. 18 (Soft Landing Augmented With Precision Location Guidance)

To soft landing capability, Configuration 18 adds location guidance capability such that impact point errors induced by midcourse or approach guidance and control inaccuracies can be guided out during the terminal descent phase. For lunar landings, impact point errors as large as 30,000 ft can be compensated for with approximately a 50% increase in ΔV capability.

The basic intelligence required for this pinpoint landing location guidance is provided by map matching techniques using correlation trackers. The immediate question and problem that arises is the matter of reconnaissance required to establish the reference target area image.

For this configuration, because of the active guidance during the terminal phase, some form of computational reference inertial frame is required. Because of this, the inertial equipment is shown to be of the complete IMU type rather than the simple IRU that was indicated for Configuration 16.

Configuration No. 19 (Beacon Aided Soft Landing)

This configuration differs from the preceding one only in that a beacon tracker, rather than a correlation tracker, is used to provide the information required for pin point terminal soft landing. This requires that an active beacon or beacon transponder be placed near the desired landing point beforehand. This in itself poses an interesting problem.

Configuration No. 20 (Manned Soft Landing)

This terminal descent configuration is applicable for pinpoint terminal soft landing for manned soft landing. The crew, making use of a gimballed sighting telescope, can provide the necessary data for generating steering commands.

3.3 Guidance System Components and Subsystems

In Section 3.2, reference was made to broad classifications of guidance system equipment. Most of these classifications contain versions which have been fully developed and qualified for space use. These are identified in the brief surveys of components and subsystems presented in this section. The surveys are not meant to be complete and all encompassing, but are included only to be indicative of the state of the art in guidance equipment development and to serve as a basis for the establishment of gross configuration characteristics.

3.3.1 IMU's

Representative inertial platform mechanizations are listed in Table 3.2. The Gemini, Apollo, Minuteman, Titan, Centaur, and Saturn platforms are conventional, high accuracy units. The IMU and associated electronics for these subsystems are in the 60 to 200 lb weight range and require from 75 to 160 watts of power. It is expected that even with the natural evolution of the state of the art, the overall weight will not be reduced substantially below 50 lb for high accuracy units.

The Nortronics A-12 platform is typical of several experimental and developmental models with an integrally mounted star tracker and does not differ significantly in weight and power characteristics from conventional systems. The two floated gimballess designs were developed for a specialized application - high g environment. This type application probably has no utility in any envisioned space mission.

The last IMU typifies an advancement in the state of the art toward miniaturization and cost reduction at the expense of accuracy. It is anticipated that an IMU of this type can be used effectively in space applications, particularly for the terminal guidance phases.

TABLE 3.2 SUMMARY OF TYPICAL STATE OF THE ART INERTIAL PLATFORM SYSTEMS

MANUFACTURER AND MODEL	PLATFORM			ELECTRONICS			COMPUTER				OPERATING ENV		COMPONENTS			ENCODER READOUT ACCURACY	GIMBAL CONFIG	MTBF (HRS)	STATUS AND REMARKS
	WT (lb)	VOL (cu.ft)	PWR (watts)	WT	VOL	PWR	TYPE	WT	VOL	PWR	TEMP	ACCEL (G's)	GYROS	ACCEL	ENCODERS				
HONEYWELL	30	.55	54	80	1.24	260	IBM	60	1.5	90	160° with coolant	10	GG8001	GG116	SYNCHROS, RESOLVERS	10 min	4	840	OPERATIONAL, GEMINI PROGRAM
AC ELECTRONICS	60	.92	72	IN PLATFORM			RAYTHEON		1.85	75		16	25IRIG	16PIPA	2 SPEED RESOLVERS	40 sec	3	4450	ADVANCED DEVELOPMENT FOR APOLLO
AUTONETICS N-17	78	2.1	90	27	1.5	250	AUTONETICS D37	43		205		14	G-6 (2DF)	16PIGA	RESOLVERS		3	---	OPERATIONAL, ADVANCED MINUTEMAN
AC ELECTRONICS	190	5.5	490	IN PLATFORM			IBM	100	2.8	210	FLUID COOLED COLDPLATE		2FBG	25PIGA	OPTISYNS		3	---	OPERATIONAL, TITAN II
NORTRONICS NIP-106	25	0.3	150	IN PLATFORM			----------				-65°F/ +125°F	200	GIK7	APE 2-10	ELECTRO-STATIC PHASE SHIFTER	6 min	FLOATED GIMBALLESS	1000	ALL ATTITUDE
HONEYWELL MH-319	30	.88	21	88	2.59	302	LIBRASCOPE	65	0.7	160	50°F/ 120°F	10	GG49	GG116	RESOLVERS		4	200	OPERATIONAL, CENTAUR
MIT FLIMBAL	75	.92	400	IN PLATFORM			UNIVAC 1824	32	.5	100			2FBG	16PIGA	PRINTED CIRCUIT	6 min	FLOATED GIMBALLESS		ALL ATTITUDE
NORTRONICS A-12	70	2	250	IN PLATFORM									SYG1000	GG177	RESOLVER		3		2 AXIS VIDICON TRACKER MOUNTED ON PLATFORM
ST-124M	150	1.9	110	55	1.4	77	IBM	75	2	138	0°F/ 120°F	10	AB5	AB3	RESOLVERS	10 sec	3		SATURN; GAS BEARING INSTRUMENTS
TELEDYNE	10	.1	85	IN PLATFORM			IN PLATFORM						TEL 2DF	TEL			4		DEVELOPMENTAL LOW COST ($25,000), MODERATE ACCURACY SYSTEM, EXTENSIVE MICROCIRCUITRY

The MTBF for a complete IMU is not expected to be signficantly more than 5000 hours. This factor, along with considerations of power conservation, indicate that for mission durations beyond about 500 hours, complete IMU shutdown will be standard operational procedure. IMU shutdown can be incorporated as an operational mode because for great portions of many space missions, the IMU functions are not required. Furthermore, for most unmanned space missions, it becomes difficult to justify the need for an IMU at all except for the launch and orbit injection phase and the pinpoint location terminal guidance phase, if such is a mission requirement. On the other hand, the gyro stabilized platform portion of the IMU, without the IMU, can serve as an excellent base for experimental equipment such as space telescope.

3.3.2 Gyros

The representative gyros listed in Table 3.3 include those gyros specifically designed for IMU application and those which can be used in strapdown applications. Not included in the table are such exotic and advanced instruments as nuclear magnetic induction gyros, laser gyros and electrostatic suspension gyros.

A significant observation to be made from the table is that the operating lifetime for single degree of freedom gas bearing gyros is an order of magnitude greater than for similar ball bearing gyros (see Honeywell GG 8001 and GG 150 units).

3.3.3 Accelerometers

Representative state of the art accelerometers are listed in Table 3.4.

TABLE 3.3 SUMMY OF TYPICAL STATE OF THE ART PLANET SCANNERS AND TRACKERS

and Model	Two Axis	Size	Weight	Power	Optical System	Type of Sensor	Detcor	Scan Speed	Spectral Range	Accuracy (instrument)	Acquisition Field of Vision	Track FOV	Instantaneous FOV	Tracking Time Constant	Operating Temp Range	Altitude (earth sensing)	Lifetime & Reliability
ATL OGO	2 heads each 2 trackers & elec box	270 in^3	13.2 lb	11 w max	Refractive telescope Plane mirror scan	Edge scan 3 points	Ge-inrsed Bolomer	20 cps dither	8–22 microns	$\pm 0.25^\circ$ at null	$\pm 45^\circ \times 2^\circ$	$\pm 2^\circ$	$1 \times 1^\circ$	50 msec	0–120°F	50–220,000 n mi	0.8632 1 year
ATL Gemini	1 sensor & elec	209 in^3	11 lb	11 w max	Refractive telescope Plane mirror scan	Edge scan 160° sector	Ge-inrsed Bolomer	20 cps dither 3 sec sector scan	8–22 microns	$\pm 0.1^\circ$ at null	$75^\circ \times 160^\circ$	$\pm 2^\circ$ dither 160° sector scan	$1.5 \times 1.5^\circ$	3 sec	0–130°F	50–2,000 n mi	MTBF = 15,000 hr
Barnes Mercury	2 heads	154 in^3	6 lb	7 w	Refractive telescope Prism scan	Conical scan 14.2 orthogonal planes	Ge-imrsed Bolomer	30 scans/sec	1.8–18	$\pm 1.0^\circ$	apex angle of cone 110°		$2 \times 8^\circ$	N/A	0–140°F	50–300 n mi	2 weeks
Barnes Electronic Scan	4 heads	250 in^3	25 lb	13.5 w	Refractive Uncorrected Schmitt	Edge scan 4 points electronic sampling	Thermile Detectr array ith 90 eleents per hed	33 scans/sec	14–40 microns	$\pm 0.5^\circ$ 1° quantization	$10 \times 81^\circ$	$10 \times 81^\circ$	$0.9 \times 10^\circ$ per detector element	0.3 sec	-40° to $60^\circ C$	15–14,600 n mi	.9 for 3 years
GE NIMBUS	2 heads 1 tracker per head	N/A	N/A	N/A	Refractive telescope Prism scan	Conical scan in parallel planes	Ge-imrsed Bolomeer	16.2 scans/sec	12–18 microns	Estimated: $\pm 1.7^\circ$ roll $\pm 0.6^\circ$ pitch	Conical Scan with 90° apex angle		$4 \times 8^\circ$	0.32 sec	32–86°F	Nominal-500 n mi circular orbit	90 days
TRW Reliable Earth Sensor	1 sensor	216 in^3	6.2 lb	5 w max	Refractive	Radiation balance	8-elemnt platinm Bolomeer array		12–18 microns	3° (3σ for 3 years)	12°	12°	12°	0.60 sec	14–140°F	6,000–19,270 n mi	0.905 3 year
Barnes Apollo Antenna Positioner	1 sensor	5" dia 6" long	3.5	1.5	Refractive	Radiation balance	4 element thermopile			1.5° at 8000mi (35° apparent earth size $.28^\circ$ at lunar distance (2° apparent earth sizes)	10° (wide FOV not spec)	10°	10°				.999989 for 14 days
Barnes "FIRM"	4 heads	500 in^3	12	20 w max		Edge scan				$.05^\circ$ ($\pm .01^\circ$ instrument error)				10 cps			
Lockheed Planet Scanner		200 in^3	5 lb	15 w max	IR-OAO plus supp Optical System					$\pm 0.5^\circ$		4.5 min					
Nortronics Lunar Horizon Tracker	4 heads	340 in^3	13 lb	10 w	4 Boowers opt. systems 20 x 40mm aper	Radiation Balance	4 Thermopiles in linear cruciform array		0.5 – 35 μ	0.1°	Hemisphere			14 msec			
Nortronics Short Range Earth Sensor		72 in^3	2.5 lb	3.5 w	Shadow mask		Photomult		S-11	$0.02 \times 0.03^\circ$	$20^\circ \times 40^\circ$						
Nortronics Long Range Earth Sensor		160 in^3	6.5 lb	65 w	50 mm f1.95 Refractive		Photomult with vibrating scanner		S-11	$\pm 0.2^\circ$	$4^\circ \times 10^\circ$						

3.3.4 Star-Sensors, Trackers and Mappers

Three basic types of star trackers are represented in Table 3.5. The single axis analog output types are generally body fixed, with a rectangular, almost slit type, field of view. The basic output desired is the offset from null, in the narrow dimension, of the image from a preselected bright star such as Canopus. This signal is used to establish a fixed S/C attitude about one axis relative to the line of sight to the chosen star.

The two axis ungimballed sensors are functionally and operationally the same as the single axis sensors above, the obvious difference being that S/C attitude control about two axes is possible.

The two axis gimballed trackers, rather than being used to control S/C attitude directly, can be used to determine the LOS (Line of Sight) direction of a star relative to the S/C axes. Two individual star sightings taken nearly simultaneously in this manner help to determine S/C attitude. IMU re-alignment following shutdown and IMU-drift calibration can also be accomplished.

Table 3.6 lists several developmental star field mappers. These devices basically can yield the information obtainable from a battery of two axis trackers. The tracker pointing and readout problems and complexity are traded for difficulties associated with map interpretation and data reduction.

TABLE 3.4 SUMMARY OF TYPICAL STATE OF THE ART ACCELEROMETERS

MANUFACTURER AND MODEL	PHYSICAL CHARACTERISTICS			CONSTRUCTION CHARACTERISTICS				OPERATING PARAMETERS			PERFORMANCE COEFFICIENTS		REMARKS
	SIZE (in)	WT (lb)	OPERATING TEMP (°F)	TYPE	PROOF MASS SUSPENSION	PICKOFF TYPE	TORQUER	RANGE (g's)	VELOCITY RESOLUTION	THRESHOLD (μg)	BIAS MAG/STAB (μg)	SCALE FACTOR STABILITY (ppm)	
MIT DESIGN 16 PIGA	2.42 x 3.92	1.8	125	PIGA	FLOTATION & MAGNETIC	E COIL	P.M. SERVO MOTOR	±30	0.12 fps				IN QUANTITY PRODUCTION
HONEYWELL GG177	1.5 x 1.8	0.3	170	TORQUE BALANCE	HINGED PENDULUM	DIFF TRANS	P.M.	±60		1	100/80	80	CENTAUR
NASA/BENDIX AB-3	3.25 x 5	2.7	104	PIGA	HYDROSTATIC GAS BEARING	MOVING COIL	P.M. SERVO MOTOR	0 To 10^+	0.15 fps	<1			SATURN
DONNER 4310	1.4 x 3 x 1.2	0.5	-40 to 200	TORQUE BALANCE	PIVOT AND JEWELL	CAPACITIVE	P.M.	20	NA	20	500/300	100	SURVEYOR
BELL MOD VII	1.75 x 1.15	<0.4	120	TORQUE BALANCE	HINGED PENDULUM	CAPACITIVE	P.M.	±2.7	NA	1	200/60	90	LEM ABORT GUIDANCE SYSTEM
AUTONETICS VM4A	2.55 x 4.8	2.3	67.5	FORCE BALANCE PENDULUM	HYDROSTATIC FLUID BEARING	CAPACITIVE	EDDY CURRENT	±20	0.12 fps				LARGE QUANTITIES PRODUCED FOR MINUTEMAN
TELEDYNE	.5 x .5 x .75	.035		FORCE BALANCE	QUARTZ FLEX	CAPACITIVE				1	/100		

TABLE 3.5 SUMMARY OF TYPICAL STATE OF THE ART STAR SENSORS AND TRACKERS

MANUFACTURER	TYPE	GIMBALING	GIMBAL READOUT ACCURACY	OPTICS	FIELD OF VIEW TOTAL	FIELD OF VIEW INSTANTANEOUS	TYPE OUTPUT	DETECTOR	SPECTRAL RESPONSE	SENSITIVITY (STAR MAG)	TRACKING ACCURACY	VOLUME	WEIGHT	POWER	STATUS AND REMARKS
BARNES ENGINEERING CO.	CANOPUS TRACKER	NONE	N/A	1 in. dia f/0.8 CASSEGRAIN	4° ROLL 30° PITCH	0.86° ROLL 11° PITCH	SINGLE AXIS ANALOG	CL-1147 RECONOTRON	S-11	+0.6 mag	± 0.1 deg	220 in.^{3}	6 lb	7.8 w	DEVELOPED FOR MARINER
BENDIX CORP. ECLIPSE-PIONEER DIVISION	OAO	± 60 deg (two axis)	20 sec	1.5 in. dia f/5.3 REFRACTIVE	1 X 1 deg	N/A	TWO AXIS ANALOG	ITT PHOTO-MULTIPLIER	S-20	+ 2.5 mag	± 9 sec	1695 in.^{3}	26.5 lb	14 w	
GENERAL ELECTRIC CO.	NASA AMES	NONE	N/A	DUAL FIELD CASSEGRAIN/ REFRACTOR	1 deg	N/A	TWO AXIS DIGITAL	ELECTROSTATIC VIDICON	0.4 - 0.7 MICRON	NOT SPECIFIED	± 4 sec	0.5 ft^3	23 lb	20 w	MULTI-TARGET CAPABILITY CLAIMED
GENERAL PRECISION INC. KEARFOTT DIVISION	MINIATURE STELLAR SENSOR	NONE	N/A	2.75 in. dia f/5.0 CATADIOPTRIC	15 X 30 min	TWIN-V SLIT RETICLE	TWO AXIS DIGITAL	SOLID STATE CdS	0.4 - 0.65 MICRON	+2.2 mag 900 foot-LAMBERTS	3 sec	49.5 in.^{3}	NOT SPECIFIED	NOT SPECIFIED	
HONEYWELL RADIATION CENTER	PASSIVE STAR SCANNER	NONE	N/A	2.75 dia f/5.0 REFRACTIVE	6 X 6 deg	V-SLIT RETICLE	?	ASCOP PHOTO-MULTIPLIER	0.34 - 0.6 MICRON	+3.0 mag	0.02 deg	840 in.^{3}	20 lb	1.4 w	FOR USE ON SPIN STABILIZED S/C. 35,000 hr MTBF
HONEYWELL RADIATION CENTER	ADVANCED STAR TRACKER	± 20° (single axis)	N/A	NOT SPECIFIED	1.5 X 1.5 deg	1.5 X 1.5 deg	TWO AXIS ANALOG	ASCOP 568A QUADRANT PHOTO-MULTIPLIER	S-11	+ 1.0 mag	± 27 sec (ROLL) ± 5 min (PITCH)	185 in.^{3}	5.5 lb	10.0 w	85,000 MTBF
ITT FEDERAL LABORATORIES	OAO SENSING HEAD	NONE	N/A	1.5 in. dia f/5.3 REFRACTIVE	1 X 1 deg	N/A		ITT PHOTO-MULTIPLIER	S-20	+ 2.5 mag	± 9 sec	155 in.^{3}	6 lb	4.5 w	
ITT FEDERAL LABORATORIES	OAO BORESIGHT TRACKER	NONE	N/A	2.85 in. dia f/1.9 REFRACTIVE	10 min	N/A	TWO AXIS ANALOG	ITT PHOTO-MULTIPLIER	S-20	+6.0 mag +4.0 mag	10 sec RMS 1.5 sec RMS	795 in.^{3}	25 lb	7.7 w	
ITT FEDERAL LABORATORIES	LUNAR ORBITER CANOPUS TRACKER	NONE	N/A	20 mm dia f/1.0	8.2 X 16 deg	1 X 16 deg	SINGLE AXIS ANALOG	ITT PHOTO-MULTIPLIER	S-20	+ 0.08 mag	± 50 sec RMS	264 in.^{3}	7 lb	8 w	
KOLLSMAN INSTRUMENT CO.	KS-137 OAO STAR TRACKER	± 43 deg	± 5 sec	1.25 in. dia f/4.0	1 X 1 deg	N/A	TWO AXIS DIGITAL	1P21 PHOTO-MULTIPLIER	S-4	+2.0 mag	± 22 sec 1σ	3.33 ft^3	45.1 lb	15.4 w	RELIABILITY = .92 1 YEAR
KOLLSMAN INSTRUMENT COMPANY	KS-177 SOLID STATE TRACKER	± 60 deg	NOT SPECIFIED	2.0 in. dia NEWTONIAN	1 X 1 deg	8 min dia	TWO AXIS DIGITAL	SILICON	0.82 MICRON PEAK	+ 2.5 mag 1000 foot LAMBERTS	60 sec RMS	1.2 ft^3	50 lb	90 w	IN DEVELOPMENT
NORTHROP NORTRONICS	NASA MSFC	NONE	N/A	2.0 in. dia f/9.0 CASSGRAIN	1 deg dia	1 deg dia	TWO AXIS ANALOG	ASCOP 568A QUADRANT PHOTO-MULTIPLIER	S-11	+ 2.5 mag	10 sec RMS	100 in.^{3} (FLIGHT MODEL)	9 lb (FLIGHT MODEL)	8 w (FLIGHT MODEL)	
NORTHROP NORTRONICS	NASA APOLLO RANGE INSTRUMENTATION	NONE	N/A	3.5 in. dia f/16.0 CASSEGRAIN	10 X 10 min	N/A	TWO AXIS ANALOG	RCA C-73496H VIDICON	0.4 - 0.6 MICRON	+ 2.5 mag 1000 foot-LAMBERTS	2 sec RMS	430 in.^{3} (FLIGHT MODEL)	16 lb (FLIGHT MODEL)	17.5 w (FLIGHT MODEL)	
SANTA BARBARA RESEARCH CENTER (HUGHES AIRCRAFT)	NASA SURVEYOR	± 15 deg	NOT SPECIFIED	REFRACTIVE	5 X 7.2	5 X 7.2	SINGLE AXIS ANALOG	1 P21 PHOTO-MULTIPLIER	S-4	- 0.6 mag	0.1 deg	NOT SPECIFIED	4.9 lb	5 w	SURVEYOR CANOPUS SENSOR

TABLE 3-6 SUMMARY OF STAR MAPPERS

MANUFACTURER	AGENCY	PROGRAM	STATUS	DETECTOR	SCANNING TECHNIQUE	MODULATION METHOD	OPTIC	FIELD OF VIEW	SCAN RATE	SENSITIVITY	VOLUME	WEIGHT	POWER
APPLIED TECHNOLOGY DIVISION AMERICAN STANDARD	NASA	ADVANCED TECHNOLOGY SATELLITE	IN DEVELOPMENT	PHOTO-MULTIPLIER	MECHANICAL (positor) 2-AXES	FREQUENCY MODULATION LINEAR 2-AXIS RETICLE	50 mm REFRACTOR	N/S	N/S	N/S	N/S	N/S	N/S
BELOCK INSTRUMENT COMPANY	NASA	APPLICATIONS TECHNOLOGY SATELLITE	IN DEVELOPMENT	SOLID-STATE CADMIUM SULFIDE	ELECTRO-LUMINESCENT PANEL	AMPLITUDE MODULATION	N/S	N/S	N/S	N/S	N/S	N/S	N/S
CONTROL DATA CORPORATION	USAF AFAL	R & D	IN DEVELOPMENT	PHOTO-MULTIPLIER	ROTATING SLIT RETICLE OR SATELLITE ROTATION	AMPLITUDE MODULATION	REFRACTOR	N/S	N/S	N/S	N/S	N/S	N/S
ITT FEDERAL LABORATORIES	N/S	"STARPAT"	EXPERIMENTAL	ITT PHOTO-MULTIPLIER	ELECTRONIC	AMPLITUDE MODULATION	2 in. dia f/1.5 REFRACTOR	$10x10^{o}$	VARI-ABLE	+ 3 mag	163 $in.^3$	7 lb	5.5 w
NORTHROP-NORTRONICS	IN-HOUSE	STAR FIELD SCANNER	DESIGN COMPLETE	ASCOP 543-D PHOTO-MULTIPLIER	ROTATING RETICLE	AMPLITUDE MODULATION	CANNON 50 mm f/0.95 REFRACTOR	45^{o} CONE	1 cps	+3.3 mag	710 $in.^3$	8 lb	11.5 w

NOTE: N/S = NOT SPECIFIED

3.3.5 Sun Sensors and Trackers

The sun sensors in Table 3.7 all operate in the 0.4 to 1.1 μ spectral region and fall into three general functional classes - sun presence sensors, null seekers, and aspect sensors. The sun presence sensor merely provides a change of state signal when the sun is within the field-of-view of the sensor and does not provide information which locates the sun within the FOV. The null seeker provides signals proportional to the angular offset of the sun's image from the sensor LOS axis. It is most commonly used as the basic sensor in a servo control system designed to point a substructure or unit of a S/C (or a particular axis of the entire S/C) towards the sun, and is rigidly fixed to the structure being controlled. A body fixed null seeker in combination with a single axis, body fixed star sensor can be used to establish a complete three axis celestial attitude reference.

The aspect sensor is merely an extension of the null seeker and is obtained by mounting the null seeker in a gimbal(s). The sun line relative to S/C axes is then determinable.

3.3.6 Planet Scanners and Trackers

The most common planet scanner listed in Table 3.8 is the earth horizon scanner, used to determine local vertical. Generally, horizon sensors operate by measuring angles to several points on the horizon or by measuring the radiance from various areas of the planet. Those sensors which measure angles to specific points on the horizon generally require some form of scan and can be broken into two categories, conical scan and edge scan. The conical scanners scan an instantaneous field of view through the planetary disk, sensing the angles to the horizon-space discontinuities. The edge scanner utilizes several separate sensors each measuring the angle to a single point on the horizon. The sensors which measure the radiance from different areas of the planetary disk are called radiation balance sensors and usually have no moving parts.

TABLE 3.7 SUMMARY OF TYPICAL STATE OF THE ART SUN SENSORS

VENDOR AND MODEL	TYPE	APPROXIMATE SIZE (in.)	WEIGHT	SENSOR	FIELD OF VIEW		PERFORMANCE			OPERATING TEMPERATURE (°C)	STATUS AND REMARKS
					ACQUISITION	TRACK	RESOLUTION	ACCURACY	LINEAR RANGE		
ADCOLE CORP. 1301	SINGLE AXIS, DIGITAL OUTPUT ASPECT SENSOR	1.2x1.8x0.5	1.5 oz.	SILICON PHOTOCELL ARRAY	$128^\circ \times 1^\circ$	$128^\circ \times 1^\circ$	1°	$.25^\circ$	NA	-70 to 100	ADCOLE CORP. DIGITAL SOLAR ASPECT SENSORS HAVE BEEN QUALIFIED FOR TWO SPACE PROGRAMS
ADCOLE CORP. 1401	TWO AXIS	2.4 x 2.4 x 0.5	3.5 oz.	"	$128^\circ \times 128^\circ$	$128^\circ \times 128^\circ$	$.5^\circ$	$.25^\circ$	NA	"	
ADCOLE CORP. 1402	"	3.3 x 3.3 x 0.8	5.1 oz.	"	$64^\circ \times 64^\circ$	$64^\circ \times 64^\circ$	$1/64^\circ$	2 min	NA	"	
BALL BROS CE-3	ANALOG, COSINE LAW OUTPUT FOR ACQUISITION	0.6D x 0.7 L	6.5 gr	SILICON SOLAR CELL	HEMISPHERICAL	NA	NA	$\pm 5^\circ$	NA	-20 to 80	CE-3 AND FE-3 CAN BE USED TO BUILD UP COMPLETE SUN SENSOR SYSTEM. BALL BROS. SUN SENSORS HAVE FLOWN ON THE OSO-1 S/C AND ARE BEING DEVELOPED FOR USE ON THE OAO AND AOSO S/C.
BALL BROS FE-3	SINGLE AXIS NULL SEEKER	0.6D x 1.1 L	6 gr	"	$\pm 10^\circ$	$\pm 10^\circ$		$\pm$ 1 min	$\pm 1^\circ$	"	
BALL BROS TE-4	SUN PRESENCE SENSOR	0.6D x 0.9 L	6 gr	"	$\pm 6^\circ$	NA	NA	NA	NA	"	
BENDIX CORP. TYPE 1818823	ANALOG, TWO AXIS NULL SEEKER	2.7D x 9L	30 oz	SILICON CELL QUAD ARRAY	$\pm 10^\circ$ (CONE)	$\pm$ 20 min (CONE)	-	5 sec (NULL)	$\pm$5 min	-55 to 50	
H.H. CONTROLS CO "REFRACTOSYN"	SINGLE AXIS NULL SEEKER	0.4 x 0.3 x 0.4	1 gr	PRISM/ SILICON CELLS	$\pm 100^\circ$			-	-	-20 to 85	
HONEYWELL-AERONAUTICAL	ANALOG, TWO AXIS NULL SEEKER	315 cu. in.	12 lb.	OPTICAL WEDGES/ SILICON CELLS	$\pm 5^\circ$	$\pm$ 20 min	-	$\pm$ 1.3 sec			BEING DEVELOPED FOR APPLICATION IN AOSO. 13 WATTS MAX POWER REQUIRED.
TRW SYSTEMS	ANALOG, TWO AXIS NULL SEEKER	3 x 4 x 4.5 6 x 4 x 4.5	2.1 lb.	FINE & COARSE SILICON ARRAYS	SPHERICAL	$\pm 17^\circ$ (CONE)	-	$\pm 0.2^\circ$ (NULL)	-	15 to 50	DESIGNED AND DEVELOPED FOR ATTITUDE CONTROL SYSTEM OF OGO
TRW SYSTEMS	SUN PRESENCE DETECTORS	4 x 2.8 x 3.5 2.8 x 3 x 2.5 4 x 2.3 x 2.5	.21 lb. .16 lb. .17 lb.	SILICON CONTROLLED RECTIFIERS	$45^\circ \times 85^\circ$ $90^\circ \times 20^\circ$ $2^\circ \times 40^\circ$	-	-	-			DESIGNED AND DEVELOPED FOR SPIN STABILIZED PIONEER. RELIABILITY = .9997 FOR 6 MONTHS IN SPACE.

NOTE: NA = NOT APPLICABLE

TABLE 3.8 SUMMARY OF TYPICAL STATE OF THE ART GYROS

MANUFACTURER & MODEL	APPROXIMATE PHYSICAL CHARACTERISTICS SIZE (in.)	WEIGHT (lb)	CONSTRUCTION CHARACTERISTICS TYPE	ROTOR BEARING	GIMBAL SUSPENSION	ANGULAR MOMENTUM (CGS)	GIMBAL FREEDOM	PICKOFF	TORQUER TORQUER	RANGE (°/hr)	SCALE FACTOR STABILITY (ppm)	LINEARITY	DRIFT g SENSITIVE MAG/STAB (°/hr/g)	g INSENSITIVE MAG/STAB (°/hr)	RANDOM OA VERT / OA HORZ (°/hr)	REMARKS
HONEYWELL GG 8001	2(dia)X3	0.7	SDF	BALL	FLOATED + PIVOT AND JEWEL	1×10^5	$\pm 1.3^\circ$	MOVING COIL	PERM MAGNET	12,750			0.36/0.03	1.3/0.03	0.008/.018	GEMINI PLATFORM, 5000 hr OPERATING LIFE
HONEYWELL GG 159	2.2 X 3	.85	SDF	GAS	HYDROSTATIC	1×10^5		MOVING COIL	PERM MAGNET	12,750			/0.08	/0.08	0.002/	50,000 hr OPERATING LIFE
MIT DESIGN 25 IRIG	2.4 X 3.7	1.5	SDF	BALL	FLOATATION + MAGNETIC	4.3×10^5	$\pm 1^\circ$	MICROSYN	MICROSYN	90,000	300	0.05%	0.2/0.2	0.3/0.2	0.07/	POLARIS, APOLLO PLATFORMS
MIT DESIGN 2 FBG-10H	2.7 X 4.5	2.9	SDF	BALL	FLOATATION + MAGNETIC	1.5×10^6	$\pm 0.5^\circ$	DUCOSYN	DUCOSYN	174			.015/.002	.015/.001		
NORTRONICS GI-K7	1.6 X 2.5	0.5	SDF	BALL	FLOATATION + PIVOT AND JEWEL	0.9×10^5	$\pm 3^\circ$	MICROSYN	PERM MAGNET	60,000	300	0.05%	1.0/-	0.5/-	0.05/	1500 hr OPERATING LIFE
SPERRY SYG-1000	1.8 X 2.8	0.7	SDF	BALL	FLOATATION + PIVOT AND JEWEL	1×10^5	$\pm 4^\circ$	MOVING COIL	PERM MAGNET	28,000	100	0.5%	0.2/0.2	0.7/0.1	0.005/0.007	2000 hr OPERATING LIFE
BENDIX AB-5	3 X 4	2	SDF	BALL	HYDROSTATIC GAS BEARING	2.6×10^6	$\pm 3^\circ$	MOVING COIL	MOVING COIL	360						SATURN PLATFORM
KEARFOTT 2542	1.4 X 2.3	0.4	SDF	BALL	FLOATATION + PIVOT AND JEWEL	0.3×10^5	$\pm 2^\circ$	U-BRIDGE AIR CORE	PERM MAGNET	72,000		0.05%	0.6/0.5	1.0/0.5	0.05/0.07	
AUTONETICS G-6	3.5 X 4	2.3	2DF	GAS	HYDRODYNAMIC			CAPACITIVE		50						FREE ROTOR, NON FLOATED
KEARFOTT 2514	2 X 3.6	1.2	SDF	BALL	FLOATATION + PIVOT AND JEWEL	10^5	$\pm 10^\circ$ (Input axis)			100,000		0.05%	1.0/0.5	1.0/0.5	0.02/0.03	SURVEYOR
TELEDYNE	2 X 1.5	0.5	2DF	GAS	FLOATATION + ELECTROSTATIC											FREE ROTOR, GIMBALLESS

Of these three types, the radiation balance sensor is the least accurate because of non-uniform IR radiation characteristics caused by clouds and day/night terminators. The radiation discontinuities at cloud edges and terminators also caused false lock ons of early versions of the scanning type earth horizon scanners because of the wide spectral sensitivity designed into the early scanners. More recent earth scanners have been designed with narrow spectral response centered near the CO_2 absorption band to prevent false lock on.

For near earth operation, the 15 μ CO_2 region appears to be optimum. The Ranger antenna positioner sensors, required to work at greater distances from the earth, are both designed to work in the visible light region. The Nortronics lunar horizon tracker has a spectral response from the visible to the far IR. The optimum spectral response regions for these latter two applications and for other planetary trackers probably have not been defined. Added research in this area should provide valuable benefits to future mission planning. Another development which would advance planetary approach guidance techniques is the development of a scanner which provides accurate stadimetric ranging data.

3.3.7 Radar

Range radars fall into two categories - these used for orbital rendezvous and those used as "altimeters". The characteristics of the cooperative rendezvous radar developed by Westinghouse for the Gemini program are listed in Table 3.9.

TABLE 3.9 COOPERATIVE RENDEZVOUS RADAR CHARACTERISTICS

	Early Version	Objective
Range rate	-100 to +500 fps	-100 to 500 fps
Maximum range	250 n.mi.	500 n.mi.
Range accuracy	0.1% or 50 ft	15 ft
Range rate accuracy	5% or 1 fps	1 fps
MTBF	3000 hours	6000 hours
Volume	1.75 ft^3	1.25 ft^3
Weight	68 lb	30 lb
Power	78 watts	23 watts
Search angle	70° solid cone	70° solid cone

Characteristics of two altitude marking radars are summarized in Table 3.10. The first is a long range device developed for Surveyor. The second is a smaller unit proposed for short range applications.

TABLE 3.10 ALTITUDE MARKING RADAR CHARACTERISTICS

	Surveyor	Short Range
Altitude mark limits	52 to 60 mi	250 to 8000 ft
Altitude accuracy	$\pm$ 0.3 mi	50 ft (3σ) at 250 ft
Power	75 watts	9 watts
Weight	10 lb	8 lb
Volume		440 in^3
Reliability		.9999 for 10 min operation

The radar altimeter and doppler velocity sensor (RADVS) used successfully on Surveyor and in development for LEM utilizes four antennas to provide slant range data and complete three axis velocity information. Pertinent characteristics of the RADVS are summarized in Table 3.11.

TABLE 3.11 RADVS CHARACTERISTICS

Slant range limits	10 to 50,000 ft
Linear velocity range	0 to 700 fps
Range accuracy	4 ft + 5%
Velocity accuracy	1 fps + 2%
Weight	33 lb
Power	590 watts

3.3.8 Laser Rangers

Laser rangers are still in the developmental stage and some basic problems will probably restrict their use to certain specialized applications. The very narrow laser beam width (which combined with the very short pulse lengths attainable makes it a highly accurate ranging and angle tracking device) makes the laser a poor search and acquisition radar. To augment the search capability, a secondary IR sensor of the earth scanner radiation balance type could be used.

The laser radar is best put to use as a ranging device. It is expected that as an altimeter, a practical laser can be developed that would operate out to nearly 100 mi with several foot absolute accuracy at all ranges.

Laser topographic mappers, have undergone some development, e.g., TRW Systems unit is capable of mapping 5 ft contour lines. In this form, this device cannot be used directly as a navigation aid but with additional logic and reference image correlation, it could be used as a map matcher for terminal guidance application.

3.3.9 Correlation Trackers

Other correlation trackers, or map matchers, are listed in Table 3.12. In addition to those listed, TRW Systems is developing an optical tracker utilizing spatial image frequency phase correlation. The expected accuracy with this device is in the range of a few feet.

Table 3.12 Map Matcher Characteristics

Technique	Company	Fundamental Mechanization	Altitude Limitations	Weather Limitations	Radiation Sensor	Correlation Time	Accuracy	Weight	Size	Input Power	Development Status
PINPOINT	Goodyear Aerospace	Active microwave scanning radar optical correlation area match	Primarily high altitude limited by field of view at low altitudes	All-weather	Active microwave radar - 9600 Mc	Approx 1 sec radar scan time and several sec optical correlation time	Estimated CEP 1700 ft at 200,000 ft altitude	Approx 100 lb	Approx 100 ft^3	Approx 1000 w	Feasibility demonstrated at low altitudes
Radiometer SIMICOR	Avco	Passive microwave scanning radar optical comparator area match	Primarily high altitude limited by field of view at low altitudes	All-weather	Passive microwave radiometer-4,000 to 16,000 Mc	Approx 1 sec radiometer scan time and several seconds of optical correlation time	Probably poorer than active microwave radar	Approx 50 lb	Approx 7 ft^3	Approx 150 w	Feasibility design completed
TERCOM	LTV	Terrain elevation profile matching digital computer correlation	Primarily low altitude limited by altitude measurement resolution at high altitudes and need for barometric altitude	All-weather	Radar altimeter and barometric altimeter	Relatively long time needed to generate terrain profile (several seconds or more) + correlation time	CEP's from 160 to 2400 ft obtained in flight tests	Approx 50 lb	Approx 5 ft^3	Approx 100 w	Flight feasibility demonstrated
TERCOR	GPL Division	Terrain elevation profile matching analog optical correlation	Primarily low altitude limited by altitude measurement resolution at high altitudes and need for barometric altitude	All-weather	Radar altimeter and barometric altimeter	Relatively long time needed to generate terrain profile (several sec or more) + correlation time	Similar to TERCOM	Approx 50 lb	Approx 5 ft^3	Approx 100 w	Feasibility design completed
Optical Area Correlator	GPL Division	Passive, nonscanning, optical lensless area correlation	Primarily high altitude limited by field of view at low altitudes	Good optical visibility over target area required	Passive optical sensor	Approx 1 sec total time to provide output data	3×10^{-3} rad LOS	19 lb	2 ft^3	30 w	Flight feasibility demonstrated at low altitudes
Microwave Correlator	GPL Division	Passive, nonscanning microwave, lensless area correlation	Primarily high altitude limited by field of view at low altitudes	All-weather	Passive microwave sensor 35,000 Mc	Approx 1 sec total time to provide output data	10×10^{-3} rad LOS	30 lb	Approx 4 ft^3	Approx 100 w	Limited feasibility design and testing completed

3.3.10 Angle Encoders

The angle encoders necessary for the determination of various tracker angles with respect to the S/C axes range in weight from 30 oz to 36 lb. From the extensive variety of available encoders, some of which are summarized in Table 3.13, the proper encoder of the desired accuracy can usually be picked for a specific application.

3.3.11 Onboard Computers and Programmer/Sequencers

For most unmanned, radio guided S/C, the onboard computer will be minimal or non-existent. Guidance and attitude reorientation commands will be generated at earth based stations and transmitted to the S/C where command receiver and decoder outputs will process the programmer/sequencer. If the unmanned S/C is required to perform any form of approach or terminal descent guidance, then a minimal type computer is required. For RADVS controlled terminal descent, the necessary computer is part of the RADVS system.

The applications for which the need for an onboard digital computer can be strongly justified are for the autonomous guidance configurations for launch and orbit injection, for certain phases of manned missions, and for missions involving rendezvous (rendezvous here is taken to apply in a broad sense and includes S/C to S/C rendezvous, planetary approach guidance, and pinpoint location soft landing). The required computer capacity depends on mission characteristics and overall operational considerations.

3.4 Gross System Characteristics

On the basis of the material presented in the preceding section, some gross system characteristics can be summarized. These are presented in the following tables, broken down into mission phases (Table 3.14, 3.15, 3.16, 3.17 and 3.18). The configuration numbers are keyed to the numbers given in Table 3.1

TABLE 3.13 SUMMARY OF TYPICAL STATE OF THE ART ANGLE ENCODERS

MANUFACTURER, TRADE NAME	PICKOFF TYPE	OUTPUT	APPROXIMATE SIZE (in.)	WEIGHT	POWER (watts)	RESOLUTION (bits)	ACCURAC
WAYNE-GEORGE, DIGISYN	CODED OPTICAL DISC	ABS	3.3 x 2.6		12	10	± 1 bit
WAYNE-GEORGE, DIGITAK	OPTICAL DISC, SINGLE CHANNEL OUTPUT	INCR	SIZE 15	3 oz.	1.8	10	± 1 bit
DATA-TECH, INCROSYN	RELUCTANCE	INCR	2 x 4.5	1.1 lb	0.9	12	±150 sec
DYNAMIC RESEARCH, OPTISYN	DUAL DISC OPTICAL INTERFERENCE	INCR	3.5 x 2.3	21 oz	1.5	12	±0.5 bit
NORDEN	MAGNETIC	ABS	3.2 x 1.75		1.0	13	± 40 sec
DATA-TECH, VERNISYN	ROTATING MAGNETIC REFERENCE SYSTEM	ABS	2.5 x 1.5		5	16	
BALDWIN	CODED OPTICAL DISC	ABS	9.9 x 4.6	36 lb	25	18	5 sec
DYNAMIC RESEARCH, THEODOSYN	DUAL DISC OPTICAL INTERFERENCE	INCR	3.4 x 2	1.3 lb	1.5	18	±5 sec
FARRAND CONTROLS, INDUCTOSYN	INDUCTANCE	INCR	3 x 0.6	10 oz	3	18	±5 sec
BALDWIN	OPTICAL	INCR	3.6 x 2	4 lb		18	±1 bit
TELECOMPUTING CORP., PHASOLVER	ELECTROSTATIC	ABS	4.5 x 1.5				5 sec
WAYNE-GEORGE, DIGISYN	PHOTOELECTRIC CELL (STROBE LAMP)	INCR	6 x 2.5	6.5 lb	25	19	±2.5 sec
NORDEN, MICROSYN	CAPACITIVE	INCR	5.3 x 3.5	5 lb	150	19	±2.5 sec
NORDEN	V-BRUSH MECHANICAL	ABS	4 x 1.75	.8 lb	0.4	19	±5 sec
NORDEN MICROGON	ELECTROSTATIC	ABS	3.5 x 2	40 oz	160	21	±1 bit

Table 3.14 Gross Characteristics of Launch and Earth Orbit Injection System Configurations

Configuration	Advantages	Disadvantages
1. Autonomous IMU	• most accurate of all configurations for launch phase but may require optical update if long coast phase involved, such as for synchronous orbit injection • calibration and alignment simple with techniques well established • can be used for many boost trajectories • well established state of the art	• heaviest, largest volume, and greatest power requirements of all configurations • MTBF not high but this is not a problem for this phase
2. Autonomous single axis platform system	• probably lighter, smaller volume, and less power than IMU • can be used for a variety of boost trajectories	• less accurate than IMU • prelaunch calibration difficult • development costs may approach IMU costs in effort to meet IMU accuracy
3. Autonomous strap-down system	• lightest, least volume and power of the three closed loop autonomous inertial systems • can be used for a variety of trajectories	• least accurate of the three closed loop autonomous inertial systems • prelaunch calibration difficult

Table 3.14 (Cont'd.)

Configuration	Advantages	Disadvantages
4. Closed loop radio guidance system	• well established state of the art • accuracy approaches attainable with IMU • lightest airborne equipment compared to preceding systems	• boost trajectory restricted to LOS from tracking stations • ground installations are expensive and costly to maintain and operate

Table 3.15 Gross Characteristics of System Configurations for Earth Orbit Corrections, Operations and Transfers

Configuration	Gross System Characteristics
5. Spin stabilized, near earth satellite with body fixed optical sensors	• onboard guidance equipment minimal • primarily useful only for small $\Delta\overline{V}$ applications because of inherent thrust vector pointing inaccuracies
6,7. Three axis stabilized near earth satellite with IRU and optical sensors	• onboard inertial attitude referencing equipment on the order of 30-35 pounds • lifetime limited by gyro life • limited to applications with $\Delta\overline{V} \lesssim 100$ fps unless lateral accelerometers added to compensate for thruster misalignment • choice between sun sensor or earth scanner dependent on mission trajectory and requirements • optical sensors can be gimballed but not necessary
8. Three axis stabilized, near earth satellite with optical sensors	• optical sensors must be gimballed • no short term memory available • potentially greater pointing accuracy than Configurations 6 and 7
9. Autonomous unaided inertial system	• long term gyro drift and power management considerations render this configuration impractical for this application
10. Radio aided inertial system	• tracking network restrictions, costs

Table 3.15 (Cont'd.)

Configuration	Gross System Characteristics
	• various modes can make system useful for manned operation, rendezvous, and for larger $\Delta\overline{V}$ application • heaviest and most complex of configurations for this phase

Table 3.16 Gross System Characteristics of Configurations for Coast and Midcourse Phases

Configuration	Gross System Characteristics
11. Spin stabilized S/C with body fixed optical sensors	• comments of Configuration 5 apply
6'. Three axis stabilized S/C with IRU and optical aids	• successfully applied to Ranger, Mariner, Surveyor • limited to moderate $\Delta\overline{V}$ application ($\cong$ 100 fps) unless lateral accelerometers added to compensate for thruster misalignment • optical sensors used to establish absolute inertial reference
10'. Three axis stabilized S/C with aided inertial system	• can be used for larger $\Delta\overline{V}$ application • heavier, larger, and more power required than Configuration 6' • operationally more flexible than Configuration 6'

Table 3.17 Gross System Characteristics of Configurations for Planetary Approach

Configuration	Gross System Characteristics
11'. Spin stabilized with body fixed optical sensors	• planetary approach trajectory can be determined with only moderate accuracy
12. Three axis stabilized S/C with IRU and optical sensors	• choice of system configuration will probably be dictated by requirements of other mission phases • trajectory determination accuracy better than for Configuration 11'.
13. Three axis stabilized S/C with aided inertial system	• choice of system configuration will probably be dictated by requirements of other mission phases • trajectory determination accuracy better than for Configuration 11'.

Table 3.18 Gross Characteristics of Configurations for Terminal Descent

Configuration	Gross System Characteristics
14. Spin stabilized S/C with marking radar	• system mechanization designed to point S/C thrust axis in desired direction prior to retro thrust phase is required • provides only open-loop pre-programmed retro thrust control and will result in semi-soft landing (approx 20 to 200 fps) • very simple system and suitable only for certain shock protected instrument packages
15. Three axis stabilized S/C with IRU and marking radar	• generally same comments as for Configuration 14
16. IRU plus doppler radar and range sensor	• makes possible controlled soft landing (10 to 20 fps) • laser ranger more accurate than radar range but added accuracy may not be necessary • all radar configuration can have array of body fixed antennas • minimal analog computer required on board • proven technique

17. V/H meter configuration	• for the same functional capabilities as Configuration 16, this configuration appears unnecessarily complex
	  • simple IRU must be replaced by full inertial system
	  • planet tracker required to point V/H meter along local vertical

Table 3.18 (Cont'd)

Configuration	Gross System Characteristics
	• either a ranging device or a range rate sensor required to unscramble V/H ratio
18. Soft landing with correlation tracker	• pinpoint landing location guidance capability added to soft landing capability • requires reference images, obtainable from previous orbiter S/C missions • for tracking type operation, fairly stringent requirements placed on accuracy with which S/C is placed in approach trajectory to insure proper tracker operation and minimum fuel consumption • correlation devices will more likely be used to update IMU at discrete intervals
19. Beacon aided soft lander	• pinpoint location guidance capability added to soft landing capability • requires accurate knowledge of beacon location (presumed to be placed by previous orbiters) • landing at beacon offset point can be accommodated • terminal trajectory constraints not as stringent as for correlation tracker configuration
20. Manned soft landing	• manual telescope sighting on desired landing area can provide required inputs for automatic pinpoint landing.

4.0 ATTITUDE CONTROL SYSTEM CONFIGURATIONS

The purpose of this section is to list representative types of attitude control configurations and indicate their applicability to particular missions or mission phases. A discussion of general system performance requirements and a means of classifying control systems will be given first as an introduction to the consideration of configurations.

4.1 Performance Requirements

The function of attitude control in space vehicles is to orient one or more spacecraft axes in prescribed reference directions in the presence of disturbances. The control system design problem associated with this function is to establish a system configuration which meets the mission requirements and constraints on performance, while optimizing the overall system in terms of some criteria. The final control configuration will thus be a function of performance requirements, expected disturbances, and the optimization criteria.

Performance requirements and the existing disturbances are linked together in establishing the direction and accuracy of the attitude orientation. The selection of a configuration depends both on the mission and on the other spacecraft subsystems such as power, communication and temperature control. The control system specifications can be described in terms of accuracy, response and dynamic range.

4.1.1 Perturbations

A major influence on control system design is the nature and magnitude of expected perturbations. The perturbations may be associated with either torque disturbances (environmental or non-environmental) or angular motion of the reference frame. The first category includes propulsion cross-coupling, crew motion, and the environmental torques due to gravitational and magnetic fields, solar radiation and aerodynamic drag. The reference frame motion consists of perturbations associated with functions such as rendezvous and docking, planetary approach and terminal maneuvers and orbital eccentricity. These operations require changing reference frames in the nature of steering control rather than maintaining a static orientation.

4.1.2 Accuracy

Primarily, the accuracy requirement provides constraints on the sensor design in terms of "near null" characteristics as well as null sensitivity and stability. In addition, system accuracy establishes the small signal requirements, such as resolution, linearity and threshold effects, on the actuators. Generally, in the absence of the other two performance requirements, the actuator and sensor requirements can be met as long as the sensor receives sufficient signal power to operate with a reasonable signal-to-noise ratio.

4.1.3 Response

Response requirements establish the transient character of the attitude motion in the presence either of significant attitude perturbations or attitude reference rotations (commanded maneuvers). The three important considerations in determining control system response requirements are related to the initial stabilization (acquisition) process, step-type disturbance torques, and the commanded maneuvers.

4.1.4 Dynamic Range

Dynamic range requirement defines the control torque range necessary to maintain the specified control capability and the sensor limits of operation at various phases of the mission. This requirement is, of course, related not only to the response requirements but also to the particular sensor-actuator configuration. The control torque requirements fall into three categories: overcoming environmental disturbance torques; maneuvering the spacecraft for initial acquisition or reorientation; and overcoming major disturbances such as midcourse thrusting.

4.2 Control System Classifications

Based on the above discussion of performance requirements, the selection of a particular control concept is dependent on pointing accuracy, control torque dynamic range, speed of response, and maneuvering requirements. A generally accepted categorization of control system configurations is based on the means of generating control torques. The general classifications are passive, semi-active and active controllers. Figure 4.1 summarizes the general classifications within these three groupings. Table 4.1 summarizes some of the characteristics of these classifications. The principles on which the control actuators operate are reviewed in Section 4.3.

4.2.1 Passive Control

Spin stabilized systems and those using environmental fields are considered passive since their normal operation expends little or no onboard energy to maintain attitude.

The simplest passive control system can be obtained by spin stabilization of the spacecraft when an axis of the spacecraft must be oriented in some inertial direction with relatively low accuracy and orientation about that axis is not required. Early spacecraft and satellites, such as Explorer and Pioneer, contained no control equipment except spin up rockets. Clearly, the system operation depended totally on having sufficient spin momentum so that expected external torques for the duration of the mission would not precess the spin axis outside the required orientation accuracy limits.

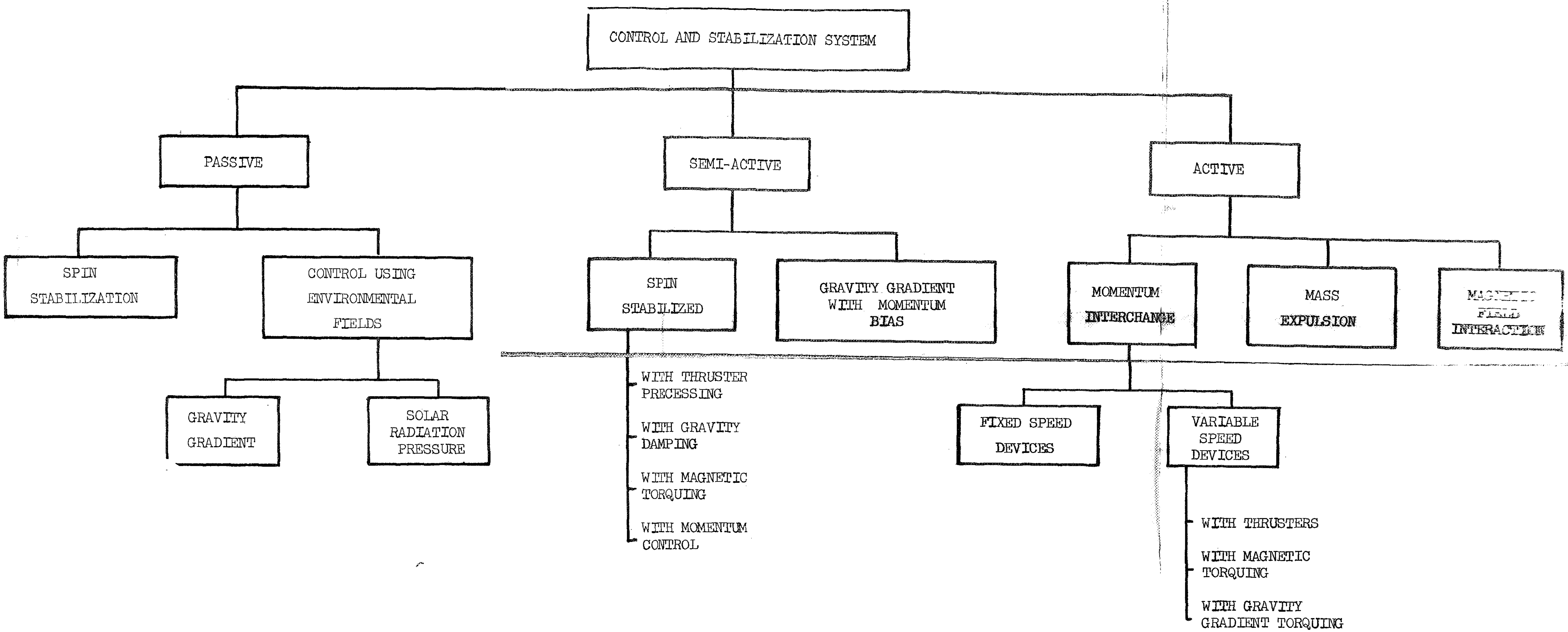

FIGURE 4.1 ATTITUDE CONTROL SYSTEM CLASSIFICATIONS

TABLE 4.1

CONTROL SYSTEM, SUMMARY OF CHARACTERISTICS

TYPE OF CONTROL		CHARACTERISTICS
Passive	General:	1) Use of environmental fields
		2) Simple
		3) Reliable
		4) Limited orientations & accuracy
		5) Low speed response
		6) No power or stored energy required
Spin Stabilization	Particular:	1) Fixed Inertial Orientation
		2) May require wobble damper or spin control
		3) Light weight
		4) Requires special provisions for optical or communication payloads
Gravity Gradient		1) Control torque a function of orbit altitude and differences in vehicle principal moments of inertial
		2) May require a passive damping device
		3) Sensitive to environmental disturbance
		4) Near circular orbit preferred
Solar Pressure		1) Control torque a function orbit orientation, spacecraft shape, emissivity and vane geometry
		2) Most applicable to interplanetary missions
		3) Light weight
		4) May require sun sensor

TABLE 4.1 (continued)

TYPE OF CONTROL		CHARACTERISTICS
Semi-Active	General:	1) Improved response and acquisition control compared to passive systems 2) Greater control authority 3) Less vulnerable to environmental torques 4) Better accuracy
Gravity Gradient with Momentum Bias	Particular:	1) Momentum storage of gyroscopic torques 2) Attitude sensor may be required
With Magnetic Torquing		1) Control torque is a function of orbital attitude and available power 2) Magnetic field sensor required
Solar Pressure with Active System		1) Active system used for acquisition or for lack of sun 2) Sun sensor may be required
Spin Stabilized with Active Precession Control		1) Active system used to precess for acquisition and station keeping 2) Attitude sensor probably required

TABLE 4.1 (continued)

TYPE OF CONTROL		CHARACTERISTICS
Active	General:	1) Widest variety of control orientation 2) Requires attitude sensor 3) Least sensitive to disturbance torques
Mass Expulsion	Particular	1) Flexibility of control configurations 2) Accuracy limited by attitude sensors and actuator resolution 3) Normally operated in a limit cycle mode
Momentum Interchange		1) Often is combined with gas systems 2) Capable of multimode operation and high precision control 3) Accuracy limited by attitude sensors 4) Requires additional torque source to relieve momentum build-up
Magnetic Field Interaction		Control torque dependent on environment, i.e., applicable to limited orbit inclinations and altitudes

The environmental field control forces include gravity gradient and solar pressure. The most common of these is gravity gradient. The solar pressure forces are less frequently used for primary control, but more often are used in combination with active devices. There are some significant operational limitations and design considerations in the use of environmental fields for control.

1) The system will have an extremely low speed of response (of the order of hours), and a limited acceleration capability (i.e., small control torques).

2) The spacecraft will, in general, be limited to having fixed nominal (equilibrium) orientation, established by the field used to generate control moments.

3) The spacecraft control authority and the orientation accuracy limitations will cause it to be sensitive to predictable and unpredictable perturbations.

4) The spacecraft will, in general, require some active control mechanism for initial stabilization.

5) Generally, a mechanism for introducing damping into the spacecraft motion must be provided.

4.2.2 Semi-Active Control

The semi-active classification is applied to systems in which the primary control is passive but is combined with an active device. Spinning satellites with thrusters for precessing and gravity gradient controlled satellites with a momentum bias and/or reaction-wheel dampers are in this category.

The addition of active control forces to a passive system is usually done for the following reasons:

1) to improve system response and dynamic range characteristics over that available with natural forces

2) to provide capability for initial acquisition or special orientation requirements

3) to improve accuracy by addition of both active torquing and appropriate sensors.

In spin stabilized spacecraft supplementary torquing is provided to enable reorientation of the spin axis from its original spin up direction. In high accuracy systems, additional control authority is used to damp out wobble of the spin axis about its required orientation. This type of wobble can be caused by unpredictable disturbance torques or changes in the direction of the true principle moment of inertia axes.

In gravity gradient systems a momentum bias (reaction wheel) may be used to improve orientation accuracy or to provide a fixed reference for damping purposes. This system should be distinguished from the active system in which the primary control is momentum interchange and gravity gradient torquing is used to remove momentum from the system. In the semi-active system, the momentum control is particularly important in damping initial rates and permitting the system to acquire normal control operation.

4.2.3 Active Control

Active attitude control systems provide the greatest design flexibility for the controlled orientation of a spacecraft in the presence of significant perturbations and maneuvers. Their control capability relative to accuracy and dynamic response is limited only by the available sensors, actuators, and the allowable system complexity. The basic concept common to active systems is the use of controlled actuation devices such as reaction wheels or mass expulsion devices.

Momentum interchange devices include fixed speed gyros and variable speed reaction wheels. The primary application for these devices has been in the area of control of periodic momenta either for maneuvering or for cyclic-disturbance torque compensation or for precision attitude control. The manner of transfer of momentum among the body axes of the spacecraft varies significantly depending on the configuration, ranging from three orthogonal fixed wheels to a single gimballed wheel. The buildup of momentum of the spacecraft is controlled by providing a counteracting torque from thrusters or an environmental field. External moments necessary for spacecraft attitude control have to date been generated primarily by thruster systems, although some systems such as Orbiting Astronomical Observatory use magnetic field torquing for momentum dumping. The major consideration in the exclusive use of mass expulsion for attitude control is that the total impulse required must be estimated and provided for prior to launch of the spacecraft. The most common propellant used is gaseous nitrogen although other gasses have been used. More efficient mass expulsion devices such as radio isotope heated gas, exploding wire, and ion engines are in developmental stages but appear to have certain limitations in terms of thrust capability, duty cycle or dynamic range. The further discussion of mass expulsion and magnetic field torquing is given in Section 4.3.

4.3 Control Configuration Mechanizations

The purpose of this section is to present a survey of possible control system configurations required for various spacecraft missions and mission phases. The control configurations presented will be described in terms of the control sensors and control orientation technique used. The various types of sensors have previously been described in Section 3.3. Therefore, as additional background to a discussion of complete control systems, a review of orientation techniques is given first.

4.3.1 Control Orientation Techniques

The control techniques under consideration will be classified into two groups, passive and active. The passive devices make use of environmental forces such as the gravity gradient and solar pressure; whereas, the active devices such as mass expulsion require direct use of onboard energy.

4.3.1.1 Spin Stabilization

The simplest possible means of obtaining control is to spin the vehicle about a known axis. If this axis is the maximum principal axis of inertia, the spin axis will remain fixed in inertial space, unless it is acted on by external disturbances. If the vehicle is initially spun about some other axis, energy dissipation will cause the vehicle orientation to change until it is spinning about the maximum principal axis of inertia. During this reorientation, however, the momentum of the vehicle remains fixed in inertial space. If it is desirable to change the orientation of the spacecraft's spin axis in inertial space in order to align it along local vertical, along the sun line, etc., the capability must be provided to activate an appropriate torquing system. A reorientation is accomplished by applying torque impulses normal to the spin axis.

The following advantages and disadvantages apply in comparing a spin stabilization with non spun.

Advantages:

1) Fixed inertial orientation with limited accuracy can be achieved with a simple passive system.

2) Accurate orientation with respect to a fixed star or slowly rotating line of sight (planet or sun) can generally be achieved with a fairly light-weight system.

3) Most disturbances including torques from thruster misalignments have only a small effect on the accuracy of a spin stabilized body, although long term torques can cause precessional drifts.

4) The system is inherently reliable due to passive operation; attitude is stable if the system fails.

5) Rotation of the spacecraft aids in thermal control.

Disadvantages:

1) No control of position about the spin axis.

2) A complex precession spin control system is required to point the spin axis along a rapidly rotating line of desired orientation.

3) Spin speed control may be required on systems where disturbance torques may cause large changes in the spin momentum of the system.

4) Passive wobble damping devices may be required.

4.3.1.2 Gravity Gradient Torques

The use of gravity gradient torques is an attractive concept because it makes use of a passive environmental force. The gravity gradient forces arise from the fact that the earth's gravitational potential varies with altitude. For this reason, the center of gravity and the center of mass of a satellite are not exactly coincident. Unless the force of gravity, applied at the center of gravity acts along a line passing through the center of mass, the resulting torque will tend to rotate the satellite. If the latter is properly configured, this torque can be usefully employed for orientation with respect to the earth's gravitational field. The nominal attitude of the vehicle is then earth referenced and limits the applications for this mode of control. For example, gravitational torques cannot, in general, be used to orient a vehicle in earth orbit toward the sun. The magnitude of torques available and the speed of response of the system is limited by the mass distribution of the vehicle and the orbital altitude. Typical capabilities of gravity gradient systems are limited to pointing accuracies exceeding $\pm .25^{\circ}$.

4.3.1.3 Magnetic Field Torques

The basic physical effect which makes possible generation of magnetic control moments is the force experienced by a moving charge in a magnetic field, $\overline{B}$. The cummulative effect of a moving distribution of charge in a magnetic field is torque, $\overline{N}$. This torque is equal to the vector cross product of the magnetic dipole moment, $\overline{m}$ and the magnetic field $\overline{B}$, i.e., $\overline{N} = \overline{m} \times \overline{B}$. For satellite altitudes greater than 100 miles above the surface, the earth's magnetic field is approximated by a simple magnetic dipole at the center of the earth. The intensity of this field is inversely proportional to the satellite altitude cubed. Using this field, a magnetic moment can be developed by simply passing current through a planar coil of wire.

This type of actuator gives limited control of the torque level. Such torque level control does indicate the need for attitude sensors to relate the coil plane to the magnetic field. In addition, the magnetic field intensity is a function of orbit position. Thus, an onboard magnetic sensor or some ground control may be required as part of the system.

In addition to their application as prime control actuators, both gravity gradient and magnetic torquing can be used in combination with a reaction wheel as a means of providing a desaturating torque to reduce the build-up of momentum in the wheel.

4.3.1.4 Mass Expulsion

The attitude control of a spacecraft with initial rates and/or in the presence of an external torque field can be simply achieved through the use of a variety of mass expulsion devices. The actuation of such devices is controlled by the output of a sensor and used to change the angular rate of the spacecraft, to keep it within some attitude error limit, or to precess the spin axis of a spin stabilized spacecraft. The mass expulsion system may either produce a torque proportional to the error signal or produce quantized torque levels for controlled periods of time to maintain the vehicle angular momentum below some prescribed limit.

A typical mass expulsion thruster system is a container of some unheated, inert gas such as nitrogen or argon with appropriate nozzles and regulators to produce thrusts. A thrust range between .001 and 1 pounds is applicable to most present unmanned spacecraft weights and lifetimes. Of course, larger thrusts may be required for very large satellites or when thrust misalignments during midcourse corrections cause excessive disturbance moments. Hot gas systems will in general have a specific impulse considerably greater than that of cold gas; however, problems associated with multiple starts and control of thrust level can limit the application of such devices.

The principal governing problem in design is the tradeoff between weight and reliability. The simplest system is a single level thrust, on-off system used to maintain the spacecraft attitude and attitude rate within certain limits. In many attitude control systems, the most efficient control scheme is a combination of mass expulsion and momentum storage, such as a momentum wheel.

4.3.1.5 Momentum Wheels

A controlled rotating inertia to provide a means of momentum interchange in a spacecraft is referred to as a momentum or reaction wheel. These devices are generally used to continuously absorb the effect of disturbance torques, to store momentum due to changes in orbital rate, and to perform special control maneuvers.

The use of reaction wheels can be understood by noting that the time rate of change of wheel momentum is equal to the applied torque (Newton's second law). In the absence of any externally applied torques, the momentum of the system remains constant with respect to inertial space. In general, there can be three body-fixed wheels whose axes may not coincide with the inertial frame. In such an instance, the wheels will change speed continuously in order to transfer momentum from one body axis to another in order that at every instant the sum of the individual momenta will equal the total constant momentum.

In the simplest case, each body axis is controlled by a single axis motor-driven flywheel. The torque can be controlled to be proportional to a given error signal. Reaction wheels are often part of a combination system as follows:

1) Reaction wheels used as temporary momentum storage to be relieved later by thruster activation when the wheel reaches saturation speed.

2) Reaction wheels used to provide damping when the gravity gradient method of obtaining control torque is used.

4.3.1.6 Control Moment Gyro

The control moment gyro is actually a variation of the momentum wheel. Using either single or two degree of freedom gyros, momentum is transferred from a fixed speed wheel to the spacecraft by electrically torquing the gyro which causes its gimbal to precess. The torquing signals are controlled by the attitude error sensor.

4.3.1.7 Solar Pressure

Bombardment of the various surfaces of the satellite by photons and protons emanating from the sun will create forces, whose magnitude and direction are determined by the reflective properties of the surfaces. The radiation power in the vicinity of the earth corresponds to a pressure of 9.4×10^{-8} lb/ft^2 for complete absorption.

There are several ways in which solar pressure can be used as a stabilizing force. In one technique, a reflective, vanelike appendage is used as solar stabilizer. The vane is mounted on the spacecraft on the side away from the Sun so that when the spacecraft has the proper attitude, the Sun's rays are parallel to the vane and no forces are generated. When the spacecraft and hence the vane are at some angle with respect to the sun's rays, a restoring force will be developed tending to return the spacecraft to the null position. For stable control the center of mass of the spacecraft must lie between the sun and the center of solar pressure on the vane. In other control systems, the vanes are actively moved according to the signal from a position sensor.

4.3.2 Control System Configurations

The classification of control systems in previous sections has been given primarily in terms of control actuators. In this section, typical control system configurations are described in terms of actuators and sensors required. The systems are arranged by mission phase to emphasize the variation of applicable configuration as a function of phases and mission requirements (see Table 4.2).

TABLE 4.2 REPRESENTATION ATTITUDE CONTROL SYSTEM CONFIGURATION PER MISSION PHASE

MISSION PHASE			LAUNCH AND EARTH ORBIT INJECTION			EARTH ORBIT CORRECTIONS, OPERATIONS, AND TRANSFERS							RENDEZVOUS AND DOCKING		COAST AND MIDCOURSE				PLANETARY APPROACH OR FLYBY		TERMINAL DESCENT	
CONFIGURATION			1	2	3	4	5	6	7	8	9	10	11	11a	6	10	12	13	13	2	2	14
TYPE OF CONTROL SYSTEM		PASSIVE			X	X	X	X							X							
		SEMI-PASSIVE							X	X							X					
		ACTIVE	X	X							X	X	X	X		X		X	X	X	X	X
GROUND-BASED RADAR, RADIO AND COMPUTER				X				X	X			X			X	X	X	X	X	X	X	
VEHICLE MOUNTED EQUIPMENT	CONTROL ORIENTATION TECHNIQUE	SPIN STABILIZATION			X	X	X		X								X					
		GRAVITY GRADIENT TORQUE						X							X							
		SOLAR PRESSURE																				
		MAGNETIC FIELD TORQUE					X															
		MASS EXPULSION							X			X	X	X		X	X	X	X			
		MOMENTUM WHEEL								X	X											
		CONTROL MOMENT GYRO																				
		THRUST VECTOR CONTROL	X	X																X	X	X
	CONTROL INPUT SENSORS	PLANET SENSOR							X	X	X											X
		SUN SENSOR							X	X	X	X				X						
		STAR SENSOR										X				X						
		IMU	X	X									X	X						X	X	
		RATE GYRO	X	X									X	X						X	X	
		RADAR/TRANSPONDER											X									
		DOPPLER RADAR																				X
		PROGRAMMER/SEQUENCER	X		X																	
		ANTENNA POLARIZATION MEASUREMENTS															X	X	X			
		OPTICAL SIGHTING												X								

Systems 1 & 2 - Three Axis Stabilization

The systems 1 & 2 represent typical attitude control configurations for the lower stages of multistage boosters. The attitude profile of the launch and injection is commanded by radio from the ground or by a programmer/computer on board. The commands are compared to the IMU attitude information; the resultant error signals are inputs to the booster autopilot. In manned missions, the final stage or payload is also three-axis stabilized.

System 3 - Spin Stabilized

For unmanned spacecraft, it is sometimes simpler to spin up the payload prior to injection. The spin axis is directed along the orbital path. The resultant angular momentum stabilizes the direction in which the injection thrust is applied and minimizes the magnitude of velocity applied normal to the required injection direction.

System 4 - Spin Stabilized

A passive spin stabilization is applicable to missions in which the orientation at injection is satisfactory for the spacecraft mission. This simplified technique has been applied to scientific satellites such as Explorer 20 which was used for radio sounding experiments. These experiments did not have high accuracy pointing requirements. Passive damping devices such as pendulums or annular rings containing a fluid mass are often used with spin stabilized to damp angular rates occurring about other axes transverse to the spin axis.

System 5 - Spin Stabilized with Magnetic Torquing

A combination of passive techniques is possible for scientific experiments requiring Earth orientation in orbit. An example is Explorer 22, in which spin rate gradually reduced to zero by magnetic despin rods. Two bar magnets mounted inside the satellite's shell were used to passively orient the spacecraft along the earth's magnetic field.

System 6 - Gravity Gradient - Radio Command

Gravity gradient attitude control can be made more versatile by using an extendible mast (a deHavilland boom) with a mass at the end. This mast can be fixed or have the capability being extended and retracted via ground commands. A radio command - gravity gradient system of this type is to be used on the first Applications Technology Satellite (ATS). Stabilization orientation accuracies of $\pm 3^{\circ}$ are expected for the ATS if circular orbit is attained.

System 7 - Two Axis Control, Spin Stabilization

A spin stabilized system can be augmented by earth sensors and mass expulsion thrusters for specialized orientations and higher precision attitude control than would normally be possible with a purely passive system. The COMSAT synchronous communication satellite is an example of such a combined system. In this system, the spacecraft is spin stabilized for injection into orbit; the spin vector is in the orbit plane. After injection, the spin vector is precessed 90° by impulses from the thrusters. It is then maintained normal to the orbit plane by error signals from the earth sensor. The earth sensor detects deviations with respect to yaw or roll depending on which part of the orbit the satellite is in. The control loop is completed by the ground tracking station which interprets the earth sensor deviations and sends commands to the on-board thrusters. This technique maintains the satellite antenna pointing to an accuracy of $\pm 1.1^{\circ}$.

System 8 - Two Axis Stabilization - Zero Net Angular Momentum

Another type of two axis control for earth orbit operation is the configuration with the spin axis earth oriented. This is applicable to earth surveillance or mapping. In this case, rotation about the spin axis is done at zero net angular momentum by balancing the output of two reaction wheels. For this type of system, the rotation about the earth pointing axis is used for thermal control and for time-sharing the earth sensor and thrusters for control about two axes rather than as spin stabilization.

An example of this type of two axis control in surveillance satellites. The satellite is spin stabilized at 120 rpm for injection into orbit, but is despun to a low rate and adjusted to zero net angular momentum for normal earth-pointing control. The system uses an earth sensor for primary control inputs and a sun sensor as a reference for time sharing of the control about the two axes normal to the spin axis. The attitude control thrusters use cold nitrogen. The thrusters maintain the system in pulsed, limit cycle condition.

System 9 - Three Axis Stabilization, Reaction Wheels, Earth and Sun Sensors

For applications employing high resolution optical equipment, full three axis stabilization is likely to be used. These systems are combination systems with normal attitude control maintained by a momentum wheel for each vehicle axis. The attitude error inputs to the momentum wheels are provided by the earth and sun sensors. Combined with the momentum wheels is a system of thrusters which are used to remove stored momentum due to secular torques; to orient the spacecraft for initial sun-earth acquisitions; and to make orbit corrections. These thrusters do not operate the system in a limit cycle as in System 8 above.

The Nimbus meteorological satellite is an example of this type of control. It is earth oriented and requires accurate pointing control for vidicon camera systems and high resolution radiometer.

For missions with very stringent pointing accuracy requirements, a dual mode version of this same control system can be used. In such a system, initial acquisition is accomplished with a fast-response, limited-accuracy system. Then, normal operational attitude control is maintained with a slow-response, high-accuracy system. This dual mode control works with two levels of torquing, two sets of reaction wheels, magnetic torquers and gas thrusters. The Orbiting Astronomical Observatory uses this type of control to attain a pointing accuracy of 0.1 arc-second.

System 10 - Mass Expulsion, Star and Sun Sensors with Radio Ground Commands

Missions with moderate accuracy requirements are suitable applications for direct control by mass expulsion. These systems normally operate with three-axis stabilization. The references for orientation are provided by a sun sensor and either an earth or star sensor. The star Canopus is frequently used due to its brightness and the fact that its declination makes it nearly normal to the ecliptic. The control functions related to mission phases and midcourse guidance corrections are supplied by radio command.

Tracking of the sun and Canopus provides heliocentric reference coordinates applicable to a variety of missions within the solar system. Both Mariner and Surveyor are representative of this type of system. This type of control system is a likely candidate for three-axis stabilized probes near the sun or to the outer planets, particularly as the complexity of scientific payloads increases and antenna pointing accuracies become higher.

System 11 - Inertial Reference Unit, Rendezvous Radar, Onboard Computer

This type of control system is typical for manned orbiting, rendezvous and docking missions. The attitude control operates in both a manual and automatic modes. The manual mode is the primary mode. The manual orbital rendezvous procedure utilizes the onboard radar to locate and track the target. The rendezvous radar furnishes range, range-rate and line-of-sight angle. This information is combined with attitude information from the inertial reference unit. The combined information is converted by an onboard computer and presented in displays of attitude and velocity to guide the crew in carrying out the maneuvers.

This type of configuration is employed for Gemini and Apollo rendezvous and docking and is applicable to logistics vehicles in support of MOL and MORL.

System 11a - Mass Expulsion, Inertial Reference, Optical Sighting, Onboard Computer

System 11a is a modification to System 11, in which an optical sighting of the object vehicle is used in place of or as a back-up to the radar transponder system.

System 12 - Spin Stabilization, Mass Expulsion, Antenna Polarization Measurements, Sun Sensors

This system is applicable to planetary approach or flyby missions and is a variation of System 6. In unmanned scientific mission, the communications antenna pointing and the performance of midcourse corrections are major considerations in the control system design. For one configuration of this type, the antenna is body-fixed pointing along the spin axis. The spin axis is then directed at the earth in the cruise attitude. If the antenna beam is directed out at an angle with respect to the spin axis, the rotation of the spacecraft generates a conical scan pattern.

The received signal contains a modulation envelope which is periodic at the spin rate. The phase and amplitude of this error signal indicates the magnitude and direction of precession required to point the spin axis at the earth. The actual precession of the spin axis is done by the mass expulsion system.

The normal procedure for making a midcourse correction is to precess the spacecraft until the thrust axis is pointed in the direction of the required velocity increment. The sun sensor (or a star sensor) is used to define a second reference direction to control the direction of precessional motion.

An alternate procedure is to make the correction with the thrust axis pointing toward the earth. By this method, two corrections on different days can provide the equivalent of a single maneuver in a variable direction. This approach adds constraints to the choice of trajectories and raises midcourse propellant requirements.

System 13 - Three Axis Control, Mass Expulsion, Antenna Polarization, Single Star or Sun Tracker

This type of active, three-axis control uses radio frequency angle tracking systems based on closed loop links with ground tracking stations. The system uses either amplitude or phase comparison of a monopulse signal for attitude control of two axes and a single star tracker for the third axis. These systems have the disadvantage of being complex and requiring large amounts of power from the ground stations.

The applicability of this approach is strongly influenced by mission trajectories, required midcourse corrections and the capability of the communication system, i.e., the RF signal transmitted by the Deep Space Instrumentation Facility. As a result, a discussion of specific modes of operation is meaningful only in terms of specific mission requirements.

System 14 - Attitude Control for Terminal Descent

The attitude control procedures related to terminal descent are closely interrelated with the guidance concepts previously discussed in Section 3 rather than existing as two separate and distinct systems. The primary attitude control requirements are a mass expulsion system to orient the spacecraft in the proper attitude for retropulsion or aerodynamic braking at the command of the guidance system and a stabilization system for use during any retrothrusting phase. The measurement of position and velocity with respect to the planet will be considered a guidance function. Accordingly, the sensors such as horizon scanners and doppler radars which make these measurements are part of the guidance system.